CR 73142

Final Report on Extensions to

FEASIBILITY STUDY FOR A MULTIPLE SATELLITE SYSTEM

Contract NAS 2-3925

Extensions No. 1 and No. 2

Prepared for

NASA
AMES RESEARCH CENTER
Moffett Field, California

Report SG 1089R-6
Volume I - SUMMARY

Space-General, a Division of
Aerojet-General Corporation

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Approved:

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ABSTRACT

The feasibility of deploying multiple satellites into a non-coplanar array for the purpose of defining spatial and temporal variations of the solar wind and the transition region in near-interplanetary space has been investigated. The initial study effort was documented by Space-General Report No. SGC 1089R-3; this report presents the results of further analyses and studies which were completed under two extensions to the basic contract.

The scope of work covered (1) further investigation of key areas in the multiple satellite system concept, including attitude control, orientation requirements, and error effects; (2) an optimization of the deployment-separation sequence, involving definition and analysis of alternate approaches other than the reference concept defined under the original contract; and (3) an evaluation of the applicability of existing satellite designs to the multiple satellite mission.

The deployment optimization study emphasized an alternate concept which involved a "pallet" (or bus) capable of lateral thrusting for altering the original launch orbit. This pallet completes orbital maneuvers between release of the four individual satellites such that the resulting satellite array forms a non-coplanar configuration which is optimum for completion of the scientific experiments. This alternate pallet design does not involve spin-off separation of the satellites; the orbit manuever capability allows achievement of desired array configurations on both the ascending and descending legs of the orbit, and the tangential separation distance along the orbit is established at a desired value and does not "grow" in the course of the system lifetime. The characteristics of this configuration are compared with that for the reference configuration defined under the initial contract. The available satellite evaluation concludes that the special balance and center of gravity requirements of either multiple satellite configuration cannot be provided by any known existing satellite configuration. However, all elements of operation and design for both of the multiple satellite system concepts have been found to be feasible within the existing technology state-of-the-art, the probable booster payload capabilities, and the desired mission implementation schedule.

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FOREWORD

This final report documents all technical work completed by Space-General under extensions to the "Feasibility Study for a Multiple Satellite System." It is submitted in partial fulfillment of the requirements of Contract NAS 2-3925, Extensions No. 1 and No. 2. The document consists of two volumes: VOLUME I - SUMMARY, and VOLUME II - APPENDICES.

The following personnel were responsible for major study tasks, and were primary contributors to the preparation of this final report:

R. L. Phen
Dr. L. Pode
E. A. Zeiner

Section 1

INTRODUCTION

The ultimate objective of the Multiple Satellite Program is the development of a system to place four spacecraft in a non-coplanar array, having a highly eccentric nominal orbit about the earth, to traverse the areas of interest - the magnetosphere, the transition region, and nearinterplanetary space. The four spacecraft will acquire magnetic and plasma data in the subsolar region which will allow the separation of time-dependent events from the motion associated with disturbances being propagated within the plasma.

The technical effort documented in this report was completed under two extensions to the basic contract NAS 2-3925, "Feasibility Study for a Multiple Satellite System." The results of the original study effort were reported in Space-General Report No. 1089R-3, which was submitted to NASA Ames Research Center in February 1967. The contents of this document, thus, refer in many cases to the work discussed in the original study contract report. Work completed under the two contract extensions may be summarized in the following areas, which make up the sections of this final report:

a. Key area analyses

b. Alternate deployment

c. Deployment comparison

d. Available satellites

Section 2

KEY AREA ANALYSES

The results of the basic study contract defined a multiple satellite system configuration which met the basic scientific experiment objectives and was determined to be feasible within the existing state-of-the-art. However, further analysis was felt to be necessary in several key areas to verify certain system parameters. This section summarizes briefly the results of these key area analyses completed under the extensions to the original contract; the detailed technical work supporting the conclusions is presented in the appendix volume.

2.1 SPIN STABILITY AND PRECESSION DAMPING

The original satellite reference design involved the deployment of a single magnetometer boom of approximately six feet in length, which resulted in a markedly asymmetric satellite configuration about the spin-stabilized roll axis. Since certain questions remained concerning the spin stabilization characteristics of such an asymmetric satellite configuration, a fairly detailed analysis of spin stability and precession damping for a single-boom satellite was completed.

The spin stability analysis for the single-boom satellite is presented in detail in Appendix I. In summary, this analysis concludes that:

a. The boom center of mass must be located on the satellite center of mass station along the spin axis, for both stowed and deployed boom configurations.

b. The major principal axis (which is the fixed spin axis after energy dissipation) can be parallel to the satellite longitudinal reference axis, for both boom-stowed and boom-deployed configurations.

c. The major principal axis will move relative to the satellite longitudinal reference axis at deployment. For the reference design configuration defined under the initial contract, this movement will be approximately 3.75 cm in magnitude.

d. The primary difference in dynamic characteristics of a single-boom satellite versus an axially symmetric configuration is in the nature of the coning motion prior to energy dissipation. As indicated in Figure 1, for the axially symmetric case the major principal axis precesses about the angular momentum vector at a fixed cone angle. (Note that the longitudinal reference axis for the satellite design may or may not correspond to the major principal axis.) For the single-boom asymmetric satellite case, the coning motion of the major principal axis about the angular momentum vector does not occur at a fixed cone angle. Rather, the major principal axis oscillates between limiting inner and outer cone angles as it precesses about the momentum vector.

e. For both symmetric and single-boom cases, energy dissipation causes the coning to decay to a steady rotation with the major principal axis along the angular momentum vector. Since sufficient precession damping capability will be included in the satellite design to assure that this coning decay occurs within a few minutes after satellite release, the asymmetric single-boom satellite configuration is entirely acceptable from a dynamic standpoint.

An analysis of the precession damping requirements for the single-boom satellite configuration was carried out to determine the required characteristics and performance of an optimum precession damper. The details of this analysis are presented as Appendix II. The results of this analysis have been summarized in Figure 2. Three types of precession dampers were considered: Type 1 - a mass constrained to move roughly parallel to the spin axis of the satellite; Type 2 - a mass constrained to move in a circular motion about the spin axis of the satellite; and Type 3 - a viscously-coupled rotor whose axis of rotation is parallel to a lateral axis of the satellite. The Type 1 and Type 2 precession dampers are found to have unfavorable location requirements for the reference multiple satellite design. The Type 3 damper is found to be the most applicable to the requirements. Its advantages include:

a. It is effective at low excitation levels.

b. The liquid rotor can completely fill the damper tube.

c. Location requirements are favorable.

d. Freedom exists in the shape of the damper loop (it need not be circular).

e. The liquid rotor can be caged by a single valve, since the tube is completely filled.

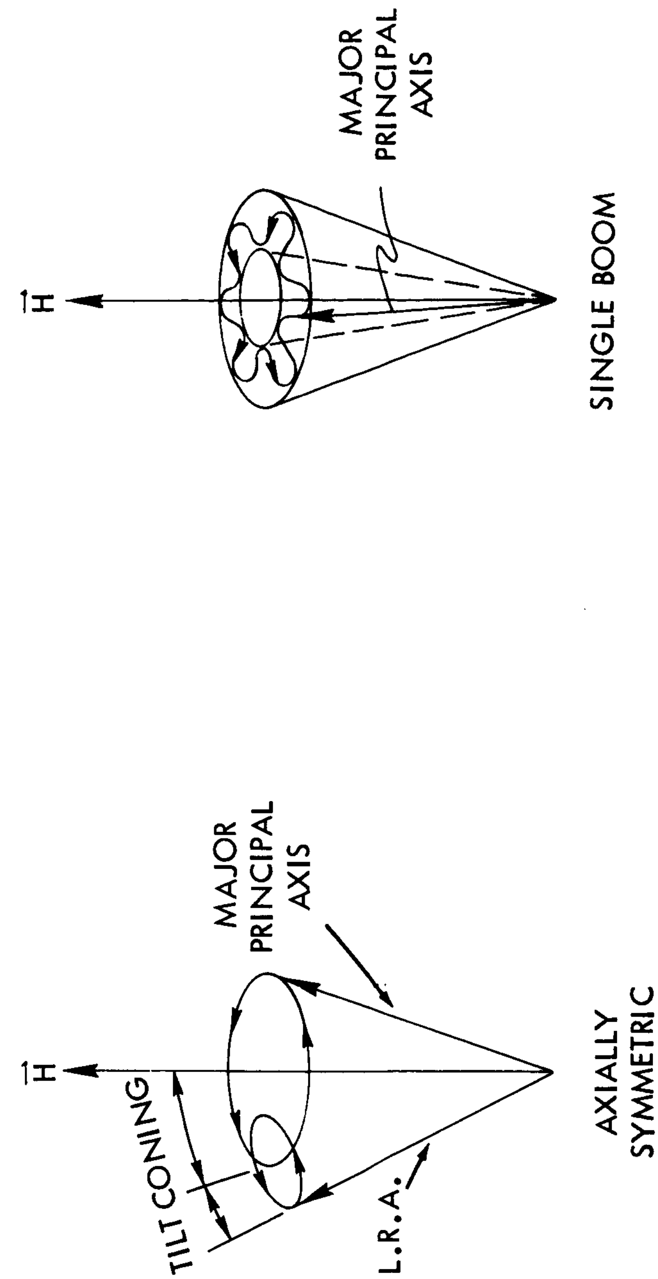
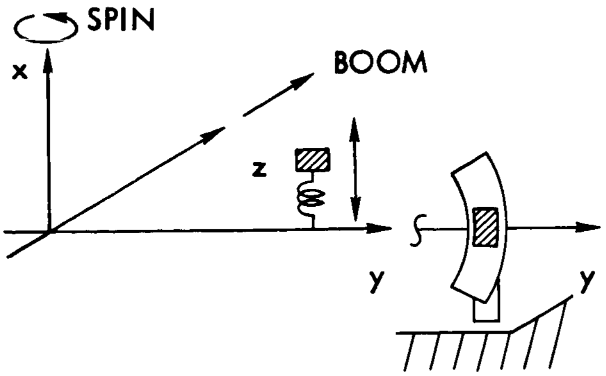


Figure 1. Comparison of Coning Motions

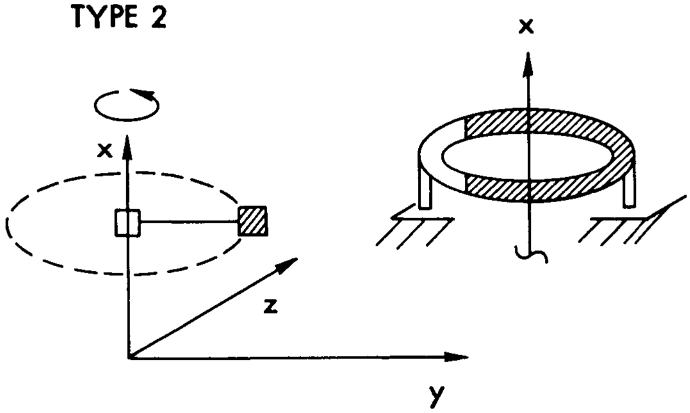
TYPE 1



LOCATION IMPORTANT FOR ONE-BOOM SATELLITE

SATURATES AT LARGE CONE ANGLES

TYPE 2



PARTIALLY FILLED TUBE — EFFECTS FINAL C.O.M.

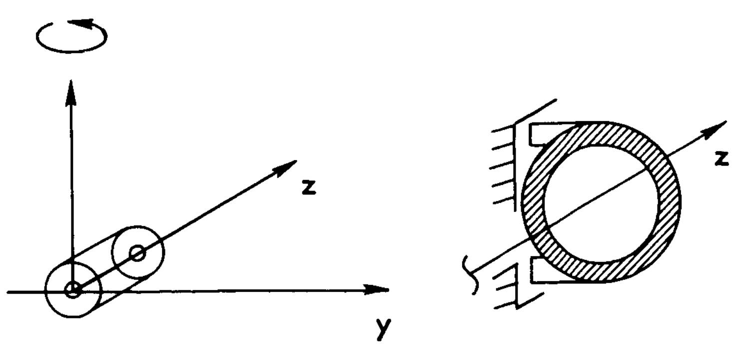
UNFAVORABLE LOCATION REQUIREMENTS

COMPLETELY FILLED TUBE

FAVORABLE LOCATION

LOOP NEED NOT BE CIRCULAR

TYPE 3



CAGED BY SINGLE VALVE

Figure 2. Alternate Precession Dampers

The application of a Type 3 rotor for the multiple satellite requirement is entirely consistent with a single-boom satellite configuration. A brief analysis also indicated that the effects of damping by a typical magnetometer boom alone are not adequate to provide the desired speed of reduction in the coning motion.

Thus, results of the spin stability and precession damping analyses of Appendix I and II verify the dynamic feasibility of the single-boom satellite configuration which was chosen as a reference design under the original contract effort.

2.2 IR ASPECT SENSING

Results of the original contract effort concluded that the use of infrared (IR) aspect sensors was most appropriate for the multiple satellite system. These sensors could provide the very accurate resolution of spin axis orientation which is necessary for the controlled spin-off separation of satellites from the pallet. Further work has been completed and is documented in Appendix III which verifies the mounting and operational feasibility of the IR aspect sensor system, and provides further substantiation of the over-all orientation sensing capability of the IR system. The results of the work presented in Appendix III may be summarized as follows:

a. Effect of the probable position uncertainties for the pallet and individual multiple satellites is small in terms of resulting errors in spin axis orientation data.

b. Mounting and field-of-view factors for the IR sensors can be selected which will provide both good orbital coverage and high accuracy for the aspect sensing system.

c. Use of dual field-of-view sensors with orthogonal viewing directions is recommended. This approach can:

    1. Resolve principal axis tilt while on the pallet through data obtained from the several satellite IR sensors.

    2. Resolve the orientation ambiguity on a quick-look basis.

    3. Guarantee single-satellite orientation data after substantial attitude drift.

It is concluded that the over-all IR aspect sensing system will provide resolution of the spin axis orientation to $< \pm 0.2^{\circ}$, which is adequate for both the pallet spin-off separations and for reduction of the satellite scientific data.

2.3 PERIGEE ALTITUDE

A more detailed study of the perigee altitude selection for the multiple satellite system was completed and is documented in Appendix IV.

The analysis included a more accurate ballistic coefficient for the multiple satellite reference design, and involved the evaluation of the effects of atmospheric density distribution. The density distribution factors considered included:

a. Day-night effects

b. Solar activity

c. Semi-annual plasma effects

d. Magnetic storms

The effects of the semi-annual plasma variations and magnetic storms were neglected in the quantitative analysis, since their magnitude was considered small relative to the other significant factors.

The effects of the lunar and solar perturbations on perigee altitude variation were analyzed, and a simplified approach suitable for trade-off studies and parametric calculations was defined. This approach is based upon a prediction considering the solar perturbation as the prime factor, with the lunar perturbation forming a secondary ripple on the major solar effect. A comparison of the prediction of perigee altitude variation, resulting period change, and lead time change versus time in orbit is presented as Figure 3. Note the agreement between results of the simple prediction approach and the variations defined by the detailed stepwise calculations of the 712 trajectory computer program.

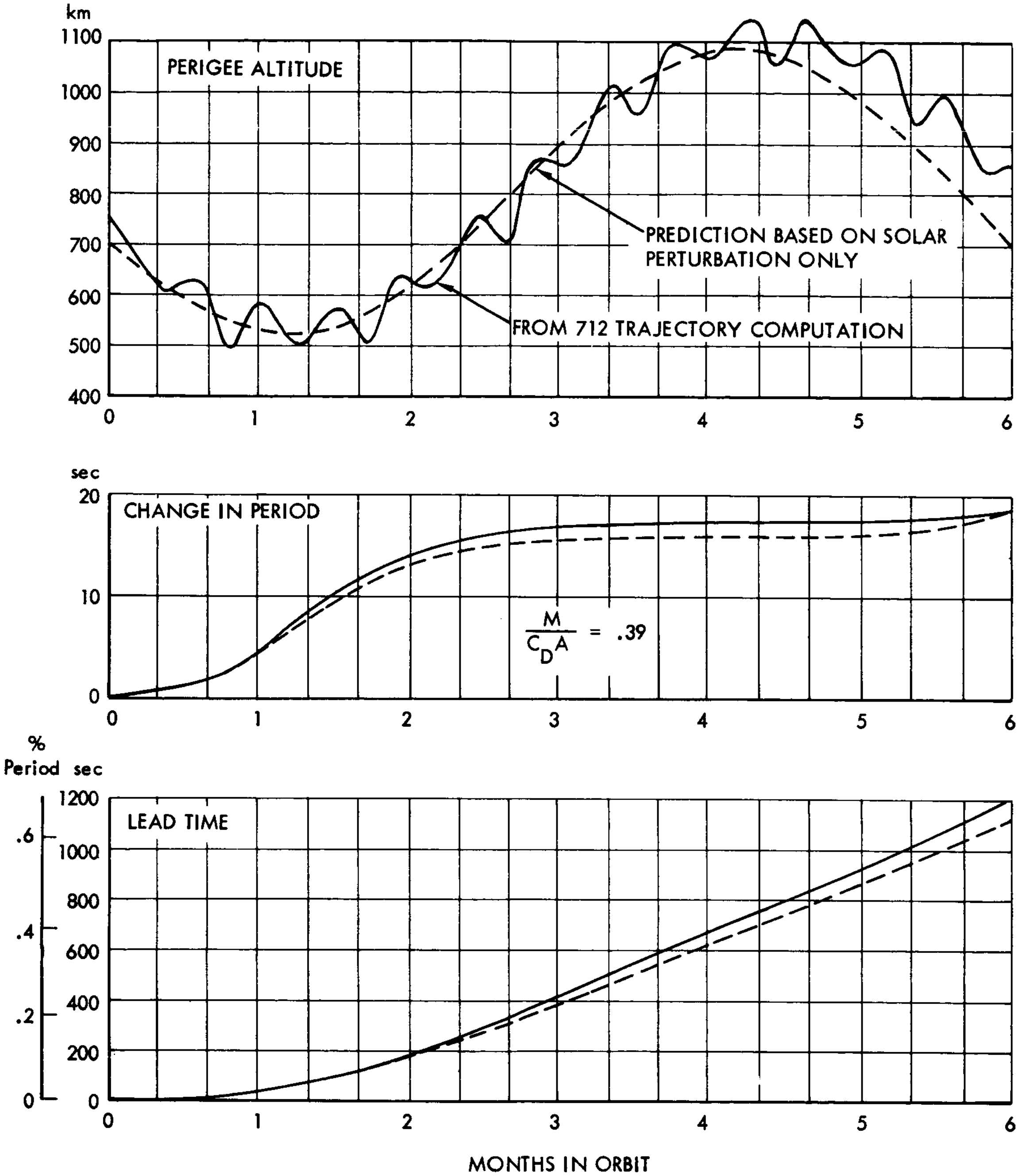


Figure 3. Comparison of Orbital Decay Computations

The required perigee altitude for the lowest satellite is defined primarily by the effects of differential atmospheric drag on tangential separation between the satellites during the course of their lifetime. Results of the detailed analysis of factors contributing to the perigee altitude effects on separation distance are presented in Figure 4. A tangential separation growth after six months of 3750 km has been selected previously as the allowable contribution due to differential atmospheric drag. If solar lead angles of up to 45° are to be allowed, Figure 4 indicates that the initial perigee altitude for the lowest satellite must be ≥ 830 km. Since the pallet orbit perigee will be some 120 km higher than that of the lowest satellite for the spin separation approach, it is concluded that the initial pallet orbit must have a perigee ≥ 950 km. It is noted that for this relatively high perigee altitude, the booster vehicle performance is dropping rapidly with increasing altitude. Thus a 50-pound payload saving can be accomplished for the mission by providing a pallet apogee kick motor, allowing the launch vehicle to achieve a low perigee orbit (e.g., 280 km) with subsequent increase in perigee to the ≥ 950 km value by use of a pallet apogee-kick velocity increment.

2.4 ORBITAL ERROR ANALYSIS PLAN

The prediction of potential variations in the multiple satellite array due to error effects is considered to be an important aspect of the system preliminary design. The system studies completed to date have in all cases considered the error effects which are judged to be of critical importance in system concept decisions. However, no attempt at a complete and comprehensive orbital error analysis has been made. It is the intent of this section to define the necessary scope of such a comprehensive error analysis, and to indicate desired results and the most reasonable approach for later completion of such an error analysis study.

The parameters of interest can be divided into three classes: (1) orbital parameters, (2) differences between orbital parameters, and (3) array characteristics. These are listed in Table 1. The orbital parameters refer to the parameters that describe the pertinent orbital characteristics of the satellites in an absolute sense, i.e., not relative to one another. Since the

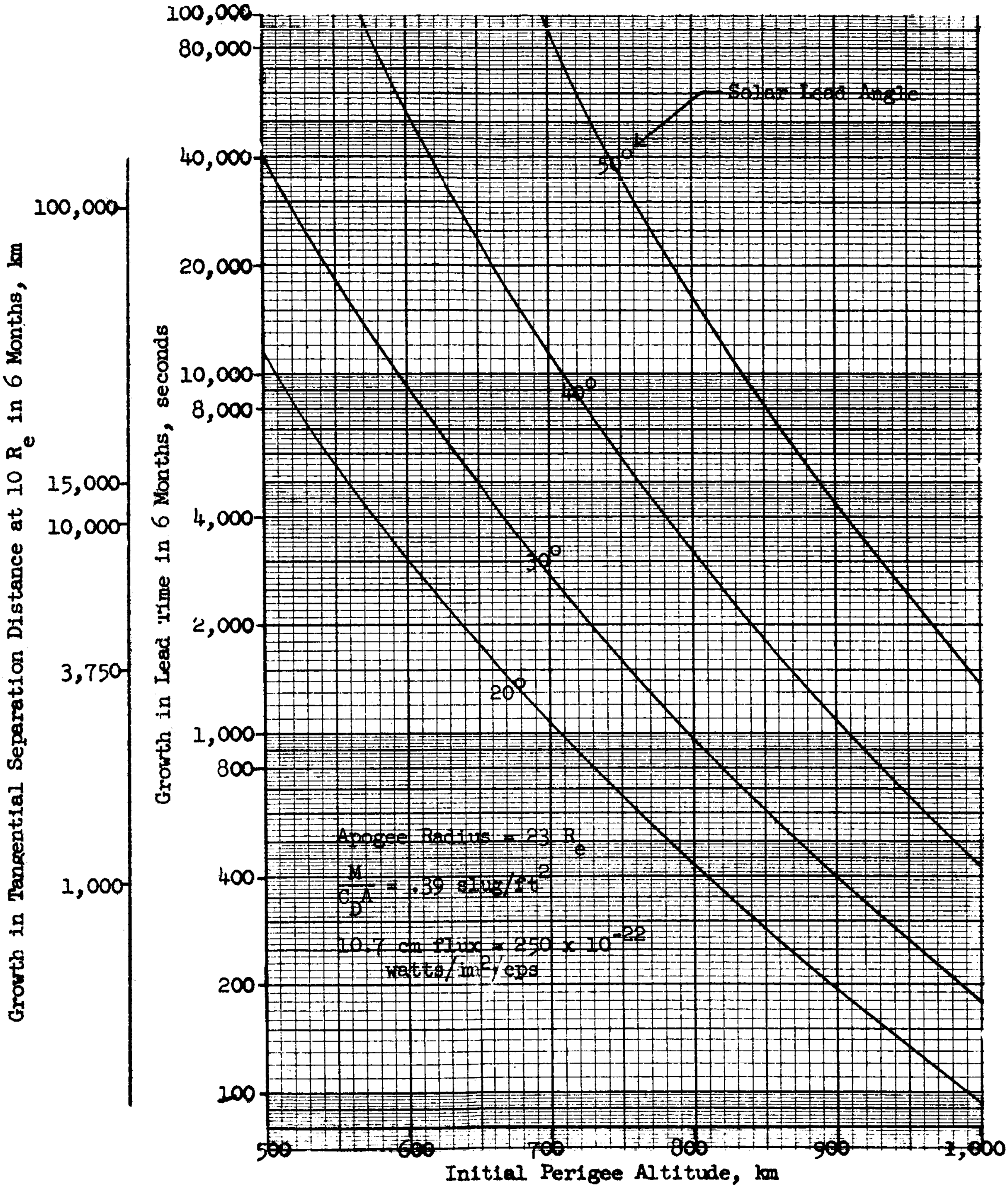


Figure 4. Lead Time Growth at Six Months

Table 1

PARAMETERS OF INTEREST

A. ORBIT PARAMETERS

Satellite 1

1. Perigee Altitude, h_p
2. Apogee Radius, r_a
3. Semi-Minor Axis, p
4. Inclination to Equatorial Plane, i
5. Argument of Perigee, ω
6. Longitude of the Ascending Node, Λ
7. Inclination to Ecliptic Plane, i_E
8. True Anomaly of Common Line, θ_c
9. Angle between Sunline and Line of Apsides, s_a

Satellite 2, 3 and 4

(Same as Satellite 1)

B. DIFFERENCES BETWEEN SATELLITES' ORBITS

Satellites 2 and 1

1. Difference in Perigee Altitude, Δh_p
2. Difference in Apogee Radius, Δr_a
3. Difference in Semi-Minor Axis, Δp
4. Difference in Period, $\Delta \tau$
5. Lead Time, Δt
6. Relative Rotation of Line of Apsides, $\Delta\Lambda$
   (In-Plane Component)
7. Relative Inclination, Δi_r
   (Angle between Orbital Planes)
8. True Anomaly of Line Common to Orbital Planes, $\Delta\theta_r$

Satellites 3 and 1, and 4 and 1

(Same as Satellites 2 and 1)

9 parameters are to be given for each of the 4 satellites, some 36 parameters are involved. The differences between orbital parameters may be expressed in terms of difference relative to the pallet's orbit, or relative to Satellite 1, depending upon the deployment scheme selected. The tables express the differences relative to the orbit of Satellite 1 giving three combinations of orbit differences: 2-1, 3-1 and 4-1. With 8 parameters describing each difference a total of 24 parameters are involved.

The description of the array characteristics will require the largest number of parameters. Although there are only five items of interest - the three orthogonal components of the separation distance, the non-coplanarity criterion, and the maximum distance - the intersatellite separation distances must be given for the three combinations of satellite differences: 2-1, 3-1, and 4-1, and must be stated for a number of points on the orbit. For the limited selection of points, some 64 parameter values are required; this doubles the total number of orbital parameters.

Not all of the orbital parameters are independent, however, In principle, 6 parameters are sufficient to specify each satellite's orbit, so that the 4 orbits could be completely specified by 24 parameters and all other parameters would be derivable therefrom. Nevertheless, evaluation of the additional parameters is necessary to permit examination of the particular items that are of interest for the Multiple Satellite mission.

The parameters enumerated above are required to describe the satellites' orbits at only one point in the operational lifetime. It is, of course, desirable that the history of the parameters and their associated errors be traced for the six-month operational lifetime. This would be done by evaluations at intervals after deployment and at critical points during the deployment, as listed in Table 2. At each point, in addition to nominal values, the three-sigma contribution of each error source, along with the RSS value of the error contributions, would be given.

Table 2

CRITICAL OPERATIONAL POINTS

- Position Error at Injection
  - Altitude
  - Down Range
  - Cross Range
- Velocity Error at Injection
  - Magnitude
  - Flight-Path Angle
  - Cross-Range Angle
- Firing of Solid Rocket to Increase Perigee Altitude
  - Magnitude of Velocity Increment
  - In-Plane Angle Error
  - Out-of-Plane Angle Error
  - Position in Orbit at Application of Velocity Increment
- Attitude Orientation Maneuver
  - Velocity Increment Applied in Out-of-Plane Direction
  - Velocity Increment Applied in In-Plane Normal Direction
  - Velocity Increment Applied in Tangential Direction
- Separation of Satellite 1
  - Magnitude of Velocity Increment
  - In-Plane Normal Velocity Increment
  - Tangential Velocity Increment
- Tangential Maneuver
  - First Velocity Increment
    - Magnitude
    - In-Plane Normal Component
    - Out-of-Plane Component
    - Time of Application
  - Second Velocity Increment
    - Magnitude
    - In-Plane Normal Component
    - Out-of-Plane Component
    - Time of Application
  - Residual Error
- Separation of Satellite 2
  - Magnitude of Velocity Increment
  - In-Plane Normal Velocity Increment
  - Tangential Velocity Increment
- Out-of-Plane Maneuver
  - Magnitude of Velocity Increment
  - Tangential Component of Velocity Increment
  - In-Plane Normal Component of Velocity Increment
  - Residual Error
- Separation of Satellite 4
  - Magnitude of Velocity Increment
  - In-Plane Normal Velocity Increment
  - Tangential Velocity Increment
- In-Plane Normal Maneuver
  - First Velocity Increment
    - Magnitude
    - Out-of-Plane Component
    - In-Plane Angle Error
  - Second Velocity Increment
    - Magnitude
    - Out-of-Plane Component
    - In-Plane Angle Error
  - Residual Error
- Orbital Perturbations
  - Atmospheric Drag
  - Solar Pressure
  - Earth Oblateness
  - Solar
  - Lunar

Table 3

ERROR SOURCES

Time	Orbital Parameters				Differences in Orbits Relative to Satellite 1		
1. Injection into Orbit	Satellites 1 to 4						
2. After Firing Solid Rocket to Increase Perigee Altitude	Satellites 1 to 4						
3. After Performing Attitude Reorientation Maneuver	Satellites 1 to 4						
4. After Separation of Satellite 1	Satellite 1	Satellites 2 to 4			Satellites 2 to 4		
5. After Completion of Tangential Separation Manuever and Separation of Satellite 2	Satellite 1	Satellite 2	Satellites 3 and 4		Satellite 2	Satellites 3 and 4	
6. After Completion of Out-of-Plane Maneuver and Separation of Satellite 3	Satellite 1	Satellite 2	Satellite 3	Satellite 4	Satellite 2	Satellite 3	Satellite 4
7. After Completion of In-Plane Normal Maneuver	Satellite 1	Satellite 2	Satellite 3	Satellite 4	Satellite 2	Satellite 3	Satellite 4
8. 30 Days After Injection	Satellite 1	Satellite 2	Satellite 3	Satellite 4	Satellite 2	Satellite 3	Satellite 4
9. 60 Days After Injection	Satellite 1	Satellite 2	Satellite 3	Satellite 4	Satellite 2	Satellite 3	Satellite 4
10. 90 Days After Injection	Satellite 1	Satellite 2	Satellite 3	Satellite 4	Satellite 2	Satellite 3	Satellite 4
11. 180 Days After Injection	Satellite 1	Satellite 2	Satellite 3	Satellite 4	Satellite 2	Satellite 3	Satellite 4

NOTE: For each time show: (1) nominal values
(2) 3σ uncertainty in value due to each error source
(3) RSS of uncertainties due to each error source

Table 3 presents an initial list of the error sources. Some items that have been included are not "error" sources in the literal sense. These items have been listed with the error sources since, for computational simplicity, it may be convenient to treat all factors that produce small changes in the orbits (other than the nominal deployment velocity increments) as though they were error sources. Also, for the most part, the listed items are not error "sources" in the sense of representing the primary origins of the errors. They represent a classification that can be used as a basis for a further breakdown which would specify primary origins. For example, an error in a velocity component may be broken down into contributions due to aspect sensor errors, thrust misalignments, inertial misalignments, etc. Such subdivision would obviously multiply the number of error sources by a substantial factor. Thus, the total number of error sources can be expected to be far greater than the nearly 50 items listed in Table 3.

The error analysis recommended would include transcription of the parameters of interest from the independent orbital parameters and the tracing of the histories of the 24 independent orbital elements. In order for the computation to be tractable each error source will be treated as independent. The interaction of critical error sources will be evaluated as necessary to understand the effects of the errors on the satellite array motion.

Section 3

ALTERNATE DEPLOYMENT

The original study contract work defined a multiple satellite system and a design configuration which generally satisfied the scientific experiment objectives and was found to be feasible from a state-of-the-art implementation standpoint. However, no comprehensive review and classification of alternate possible configurations was completed due to time and budgetary limitations of the original effort. Although the performance obtained with the original reference multiple satellite system appeared to be acceptable, it is possible that improvements in some characteristics could be obtained with alternative approaches. Thus, a study of the alternate possibilities for satellite separation and deployment was completed and the most promising alternate approach was selected, analyzed in further detail, and compared with the original reference system. This section presents the results of the alternate deployment study.

3.1 OBJECTIVES

The major deficiencies of the original reference multiple satellite deployment scheme are:

a. Large in-plane normal separation distances are provided only on the descending leg of the orbit. The smaller in-plane-normal separation distances provided on the ascending leg result in an array that is much closer to being planar. Hence, the ascending leg (i.e., half of the orbit) is of less value for the scientific experiment purposes.

b. The maximum in-plane normal separation distances that are provided by the current scheme are in the order of 800 km, being limited by the velocity increment that can be obtained by spin-off. Larger distances, in the order of 1,500 km, would be preferable.

c. For the current deployment scheme the growth of tangential separation distance between the members of one of the satellite pairs is necessary in order to obtain non-coplanarity. This requires the deliberate off-setting of the out-of-plane velocity increments from the normal to the orbit in order to be assured of the presence of a tangential velocity increment component. Although some tangential components will be present inadvertently, the deliberate imposition of a tangential component is undesirable since it decreases the allowable margin of error. The attainment of non-coplanarity without the growth of tangential separation distance would allow increased accuracy in deployment to be pursued without restriction.

In examining alternative deployment schemes the main objective is to determine to what extent these deficiencies might be remedied without degrading other aspects of the system's performance or aggravating design problems.

3.2 ASSUMPTIONS AND CONSTRAINTS

Present considerations will be restricted to alternative deployment schemes that are applicable to Mission Mode 3. With this mission mode the satellites are not individually equipped with attitude reorientation systems. Hence, the attitude reorientation must be completed prior to the deployment of the satellites. The satellites are, therefore, deployed from a pallet that is spinning about an axis that is very closely aligned with the normal to the orbit.

It is assumed that the satellites are to be nominally identical, that they are mounted on the pallet with their roll axes parallel with the pallet's roll axis, and that they are mounted in a configuration that is a symmetrical with respect to the pallet's roll axis. Thus, if one satellite is mounted with its center of mass laterally displaced from the pallet's roll axis, it must be balanced by another satellite that has its center of mass displaced in the opposite lateral direction.

When a laterally-mounted satellite is separated from the pallet it is spun off, retaining its circumferential velocity. To avoid an unsymmetrical configuration for the elements remaining on the pallet, which could interfere with the attainment of a clean separation, it is assumed that any spin-off separation will be restricted to the separation of symmetrical elements. Accordingly, each satellite must either be mounted: (a) with its center of mass along the pallet's roll axis, in which case it will be separated axially; or (b) with its center of mass displaced laterally, in which case it is separated by the pallet by spin-off simultaneously with the spin-off separation of its opposite member. Spin-off separation can involve either the separation of individual satellites or the separation of satellite-pairs, i.e., two satellites joined in a dumbbell-like assembly. In the latter case, the pair assemblies will be parted subsequent to separation from the pallet.

3.3 CLASSIFICATION

With the restrictions set forth above, the possibilities with respect to separation from the pallet can be classified as follows:

I. Spin-off separation of satellite pairs; pairs mounted on pallet with dumbbell axis parallel to the pallet's roll axis.

II. Spin-off separation of satellite pairs; pairs mounted on pallet with dumbbell axis perpendicular to the pallet's roll axis.

III. Spin-off separation of all satellites at one time.

IV. Two-stage spin-off separation; at each stage two satellites are spun-off.

V. Spin-off separation of two satellites; axial separation of the other two.

VI. Axial separation of all four satellites.

Figures 5 through 10 present the general mounting arrangements and the separation sequences associated with each of these schemes.

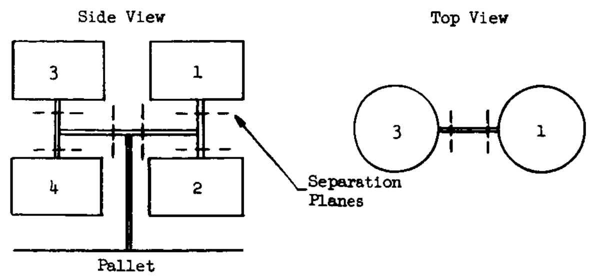
3.4 DEPLOYMENT SCHEME I - SPIN-OFF SEPARATION OF SATELLITE PAIRS; PAIRS MOUNTED ON THE PALLET WITH DUMBBELL AXIS PARALLEL TO THE PALLET'S ROLL AXIS

This approach has been investigated in some detail since it is the one corresponding to the reference design. Its main advantage is that it involves only one spin-off separation and imposes minimum burdens on the pallet. However, since only one in-plane separation is obtained, large in-plane normal separation distances cannot be obtained on both legs of the orbit and non-coplanarity is dependent upon growth of tangential separation.

Deployment Scheme I

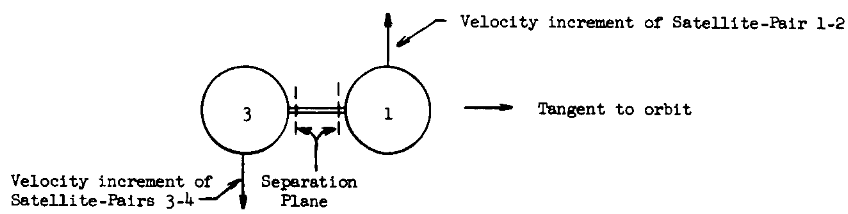
Spin-Off Separation of Satellite Pairs; Pairs Mounted on Pallet
With Dumbell Axis Parallel to the Pallet's Roll Axis

General Mounting Arrangement

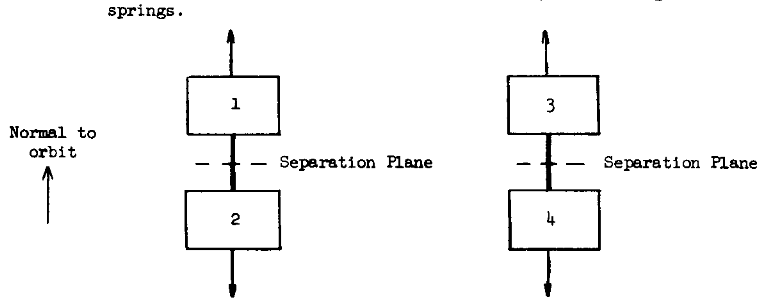


Separation Sequence

Step 1: Separate satellite pairs from pallet by spin-off with differential velocity increment in the in-plane normal direction.



Step 2: Part satellite pairs axial with small velocity increment generated by separation springs.



Step 3: Fire axial rockets to obtain out-of-plane velocity increments.

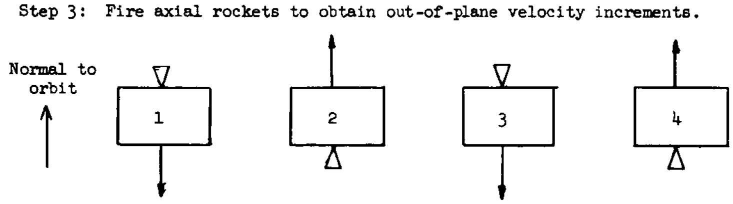
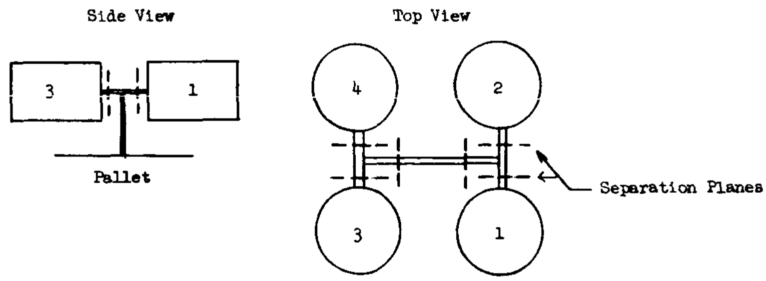


Figure 5. General Mounting Arrangement and Separation Sequence for Deployment Scheme I

Deployment Scheme II

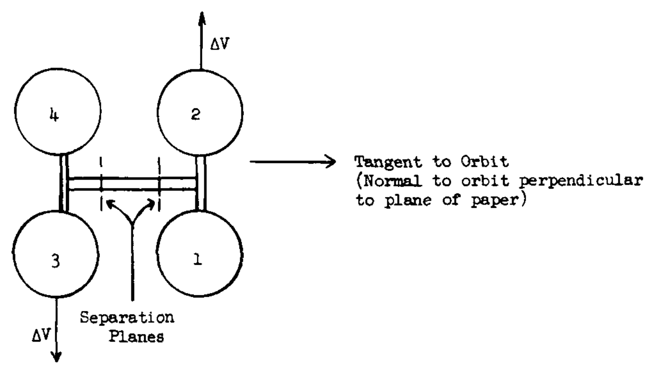
Spin-Off Separation of Satellite Pairs; Pairs Mounted on Pallet
With Dumbell Axis Perpendicular to the Pallet's Roll Axis

General Mounting Arrangement

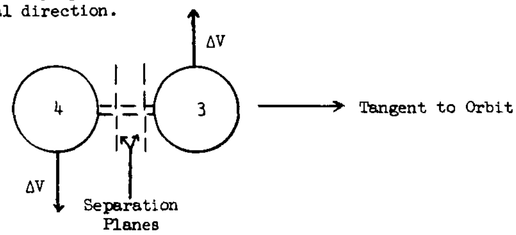


Separation Sequence

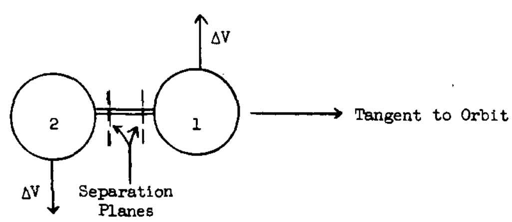
Step 1: Separate satellite pairs from pallet by spin-off with differential velocity increment in the in-plane normal direction.



Step 2: Part Satellite-Pair 3-4 by spin-off separation with differential velocity increment in the in-plane normal direction.



Step 3: Part Satellite Pair 1-2 by spin-off separation.



Step 4: Fire axial rockets to obtain out-of-plane velocity increments.

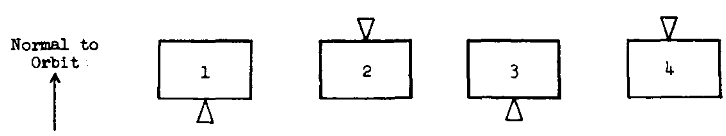


Figure 6. General Mounting Arrangement and Separation Sequence for Deployment Scheme II

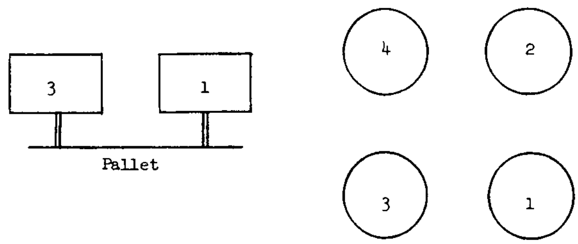
Deployment Scheme III

Spin-Off Separation of all Satellites at one Time

General Mounting Arrangement

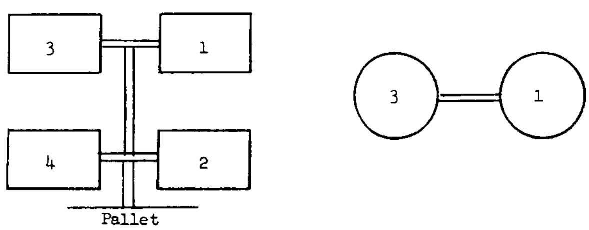
Alternative A

Side View | Top View



Alternative B

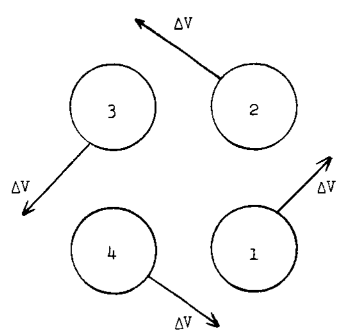
Side View | Top View



Separation Sequence

Alternative A

Step 1: Separate all four satellites by spin-off.



This alternative is not acceptable since a large tangential velocity component cannot be avoided.

Alternative B

Similar to Deployment Scheme I except that structural interconnections between departing satellites (dumbbell bars), are deleted.

Figure 7. General Mounting Arrangement and Separation Sequence for Deployment Scheme III

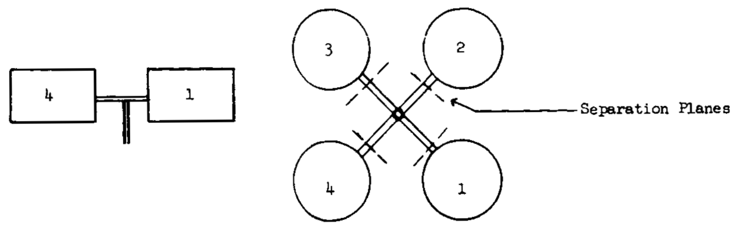
Deployment Scheme IV

Two Stage Spin-Off Separation; At Each Stage Two Satellites are Spun-Off

General Mounting Arrangement

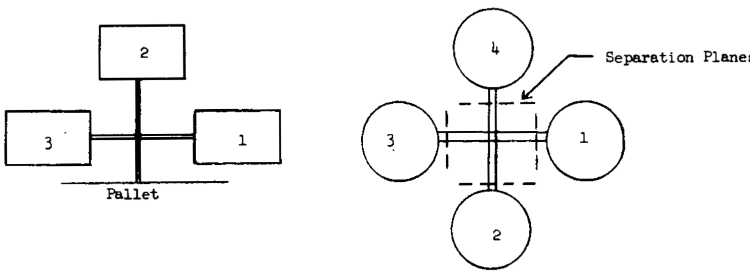
Alternative A

Side View | Top View



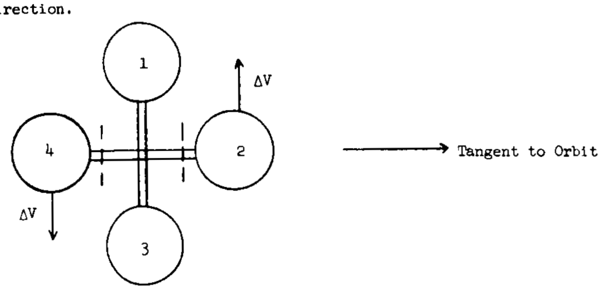
Alternative B

Side View | Top View



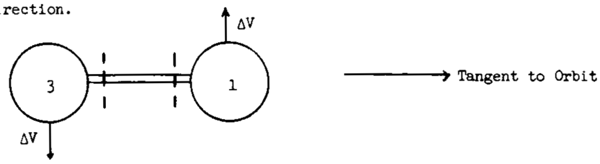
Separation Sequence (Applicable to either mounting alternative)

Step 1: Separate Satellites 2 and 4 by spin-off with differential velocity increments in the in-plane normal direction.



Step 2: Perform Pallet Orbit Change Maneuver (optional)

Step 3: Separate Satellites 1 and 2 by spin-off with differential velocity increments in the in-plane normal direction.



Step 4: Fire axial rockets to obtain out-of-plane velocity increments. (This step might be deleted if Step 2 is used to obtain out-of-plane velocity increments.)

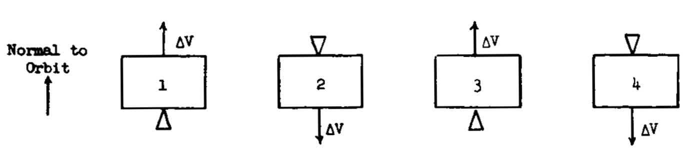
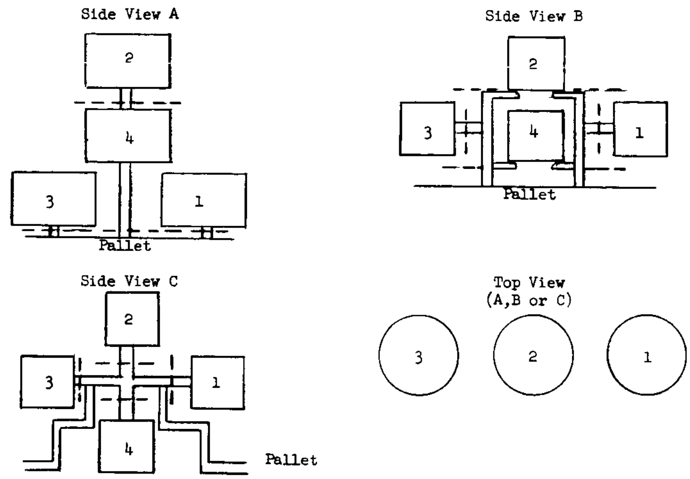


Figure 8. General Mounting Arrangement and Separation Sequence for Deployment Scheme IV

Deployment Scheme V

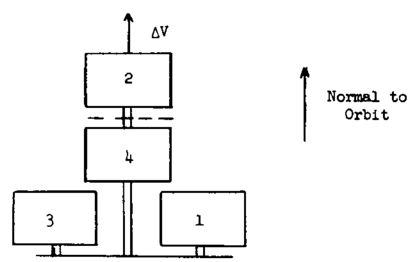
SPIN-OFF SEPARATION OF TWO SATELLITES; AXIAL SEPARATION OF THE OTHER TWO

General Mounting Arrangement

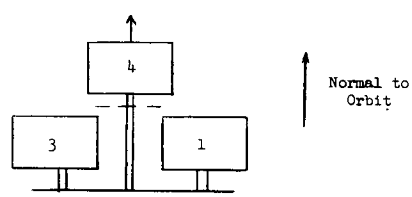


Separation Sequence

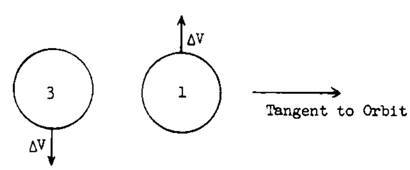
Step 1: Separate Satellite 2 axially with small velocity increment generated by reaction springs.



Step 2: Perform pallet orbital maneuver. Then separate Satellite 4 axially with small velocity increment generated by separation springs.



Step 3: Perform pallet orbital maneuver. Then separate Satellites 1 and 3 by spin-off with differential velocity increments in the in-plane-normal direction.



(Note: Steps 2 and 3 can be reversed, mounting arrangement permitting.)

Step 4: Fire Satellites' axial rockets

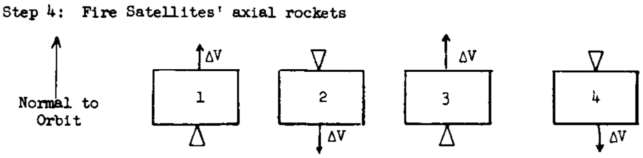
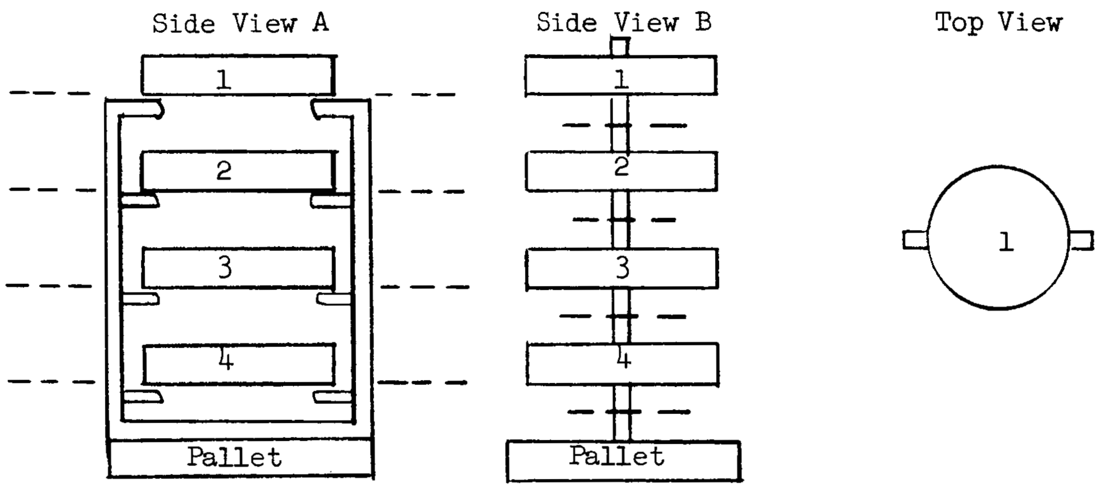


Figure 9. General Mounting Arrangement and Separation Sequence for Deployment Scheme V

Deployment Scheme VI

AXIAL SEPARATION OF ALL FOUR SATELLITES

General Mounting Arrangement



Separation Sequence

Step 1: Separate Satellite 1 axially with small ΔV from reaction springs.

Step 2: Perform pallet orbital maneuver to obtain tangential separation distance. Separate Satellite 2 axially with small ΔV from reaction springs.

Step 3: Perform pallet orbital maneuver to obtain out-of-plane separation distance. Separate Satellite 3 axially with small ΔV from reaction springs.

Step 4: Perform pallet orbital maneuver to obtain in-plane normal separation distance. Separate Satellite 4 axially with small velocity increment obtained from reaction springs.

Figure 10. General Mounting Arrangement and Separation Sequence for Deployment Scheme VI

It will be noted that for some of the other deployment schemes these shortcomings are relieved by using the pallet to perform orbital maneuvers, in addition to its attitude reorientation maneuver. Deployment Scheme I does not lend itself well to this approach since it would require the use of two pallet systems, one pallet system accompanying each of the satellite-pairs. This comment is also applicable to Deployment Scheme II. For Deployment Scheme III, in which all the satellites are separated at one time, pallet orbital maneuvers are totally inapplicable. Such maneuvers are applicable to Deployment Schemes IV, V and VI (especially to the latter) in which some of the satellites are retained on the pallet after one or more of the satellites have been separated.

Although perhaps not the best possibility with respect to structural design, the mounting arrangement for Deployment Scheme I is not unreasonable. However, the attainment of a favorable moment-of-inertia ratio proved to be difficult for the pallet and was impracticable for the satellite pairs, requiring limited duration in the pair configuration.

3.5 DEPLOYMENT SCHEME II - SPIN-OFF SEPARATION OF SATELLITE PAIRS; PAIRS MOUNTED ON PALLET WITH DUMBBELL AXIS PERPENDICULAR TO THE PALLET'S ROLL AXIS

In this case, upon separation from the pallet the satellite pairs spin about an axis that is perpendicular to the dumbbell axis. Hence, the parting of the pairs results in secondary spin-off separations, resulting in a total of three spin-off separations. This multiplicity of spin-off separations can be used to increase the maximum in-plane normal separation distance obtained and to obtain large in-plane normal separation distances on both legs of the orbit. Consider, for example, the following timing for the separation sequence:

1. Satellite Pairs 1-2 and 3-4 are spun-off from the pallet at about 10 hours after perigee passage on the ascending leg; Pair 1-2 going inward and Pair 3-4 going outward.

2. Satellite Pair 3-4 is parted by spin-off separation at about 10 hours prior to next perigee passage on the descending leg; Satellite 4 going inward and Satellite 3 going outward.

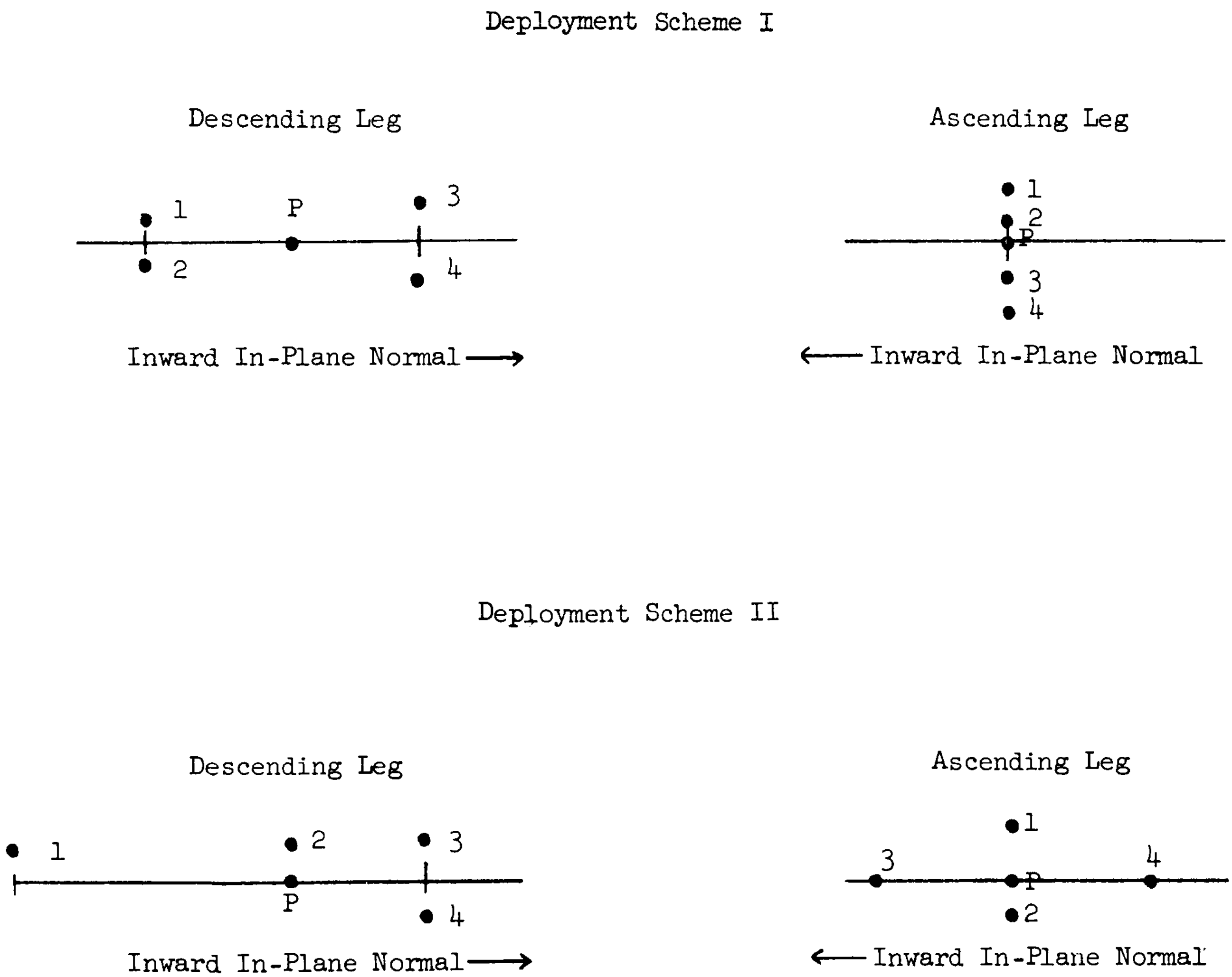
3. Satellite Pair 1-2 is parted by spin-off separation at about 10 hours after this perigee passage; Satellite 1 going inward and Satellite 2 going outward.

(The subsequent out-of-plane velocity increments provide the additional separation distance necessary to obtain a non-coplanar array.)

It can be expected that, for each of the three spin-off separations, the distance between the centers of mass of the separating elements will be about the same as for the one spin-off separation involved in Deployment Scheme I. Hence, the differential velocity increments obtained with each spin-off would be equal to that obtained in Deployment Scheme I and, as indicated in Figure 11, the foregoing deployment sequence will produce an in-plane normal separation distance on the descending leg of the orbit that is about 1.5 times as great as that obtained with Deployment Scheme 1; while on the ascending leg of the orbit an in-plane normal separation distance is produced that is equal to that obtained on the descending leg with Deployment Scheme I.

This improvement in in-plane normal separation distances is not obtained without some concomitant disadvantages. It would be necessary to execute three spin-off separations and, because of the multiplicity of spin-offs and the superposition of velocity increments, the attainment of the required accuracy would be more difficult. Also, along with the larger in-plane normal separation distance, a larger difference in perigee altitude would be introduced. This would require that the perigee altitude of the pallet's orbit at injection be raised by about 60 km. On the other hand, the "four-on-the-floor" mounting arrangement that can be used with this scheme is desirable because a favorable moment of inertia ratio is readily obtained for all configurations, i.e., the pallet, the satellite-pairs and the satellites. The mounting arrangement is also satisfactory from a structural viewpoint.

This scheme provides no benefit with respect to the requirement for growth of tangential separation to obtain a non-coplanar array.



P indicates pallet

1, 2, 3, 4 indicate satellite reference number

Figure 11. Comparison of In-Plane Normal Separation Distance Obtainable with Deployment Schemes I and II

3.6 DEPLOYMENT SCHEME III - SPIN-OFF SEPARATION OF ALL SATELLITES AT ONE TIME

This scheme does not appear to offer any significant advantages relative to the others. The "four-on-the-floor" mounting arrangement (alternative A), is not feasible since the four velocity increments obtained by spin-off separation would then be separated by 90 degrees and it would be impossible to avoid an excessive tangential component. The two-up/two-down mounting arrangement (Alternative B) would yield spin separation velocity increments that are nominally identical to those obtained with Deployment Scheme I. The only difference from Deployment Scheme I is the absence of the dumbbell bar. Deletion of the dumbbell bar offers no significant advantage and could introduce a hazard of collision between satellites shortly after separation.

3.7 DEPLOYMENT SCHEME IV - TWO STAGE SPIN-OFF SEPARATION; AT EACH STAGE TWO SATELLITES ARE SPUN-OFF

Two mounting arrangements are feasible with this scheme. These are the "four-on-the-floor" arrangement (Alternative A), and the two-up/two-down arrangement with the upper pair at right angles to the lower pair (Alternative B). The latter arrangement may be desirable for providing (1) convenient mounting of pallet equipment in the space below the upper pair, (2) more room for boom stowage, (3) allowing the design of satellites of greater diameter, and (4) for facilitating the retention of a favorable moment-of-inertia ratio for the pallet after the separation of the first two satellites.

The choice of mounting arrangement has no direct consequence on the separation sequence, which for this scheme involves a double spin-off. The double spin-off allows large in-plane normal separation distances to be obtained on both legs of the orbit. For example, spin-off separation of Satellites 2 and 4 at about 10 hours after perigee passage would provide significant in-plane-normal separation distances on the descending leg; while spin-off separation of Satellites 1 and 3 at about 10 hours prior to perigee passage would provide significant in-plane normal separation distances on the ascending leg. Presumably, the effective spin-off arm, i.e., the distance between the centers of mass of the departing elements, would be about the same as for Deployment

Scheme I. Hence, Scheme IV would provide in-plane normal separation distances on both legs of the orbit of a magnitude which Scheme I provides on the descending leg only.

Since two of the satellites are retained on the pallet after the separation of the first two, Scheme IV, provides an opportunity to aid the deployment through the execution of an orbital maneuver by the pallet between the two spin-off separations. This maneuver could be used to provide (1) tangential, and/or (2) additional in-plane normal, and/or (3) out-of-plane separation distances. Tangential separation distance could most readily be obtained in the following manner: when the pallet next passes near perigee, after separation of the first two satellites, a tangential velocity increment is applied to the pallet using a pulsed laterally-thrusting jet. Assuming for definiteness that this velocity increment is in the direction opposite to the velocity vector, it will result in a decrease in the pallet's orbital period. The pallet is then allowed to coast, say, for one orbit. Near the next perigee passage a velocity increment of equal magnitude but opposite direction is applied. This restores the orbital period to its original value. However, during the coast a lead time has been built up. A tangential velocity increment in the order of .8 meters/sec would decrease the orbital period by about .4 percent. Hence, in one orbit a lead time of about .4 percent of an orbital period would be obtained. This would provide tangential separation distances in the section of the orbit of most interest, in the order of 1,500 km, without a subsequent growth of tangential separation distance.

Additional in-plane-normal separation distances could be produced by using the pulsed laterally-thrusting jet to obtain an in-plane normal velocity increment. This like the spin-off separation, is most effectively done at large distances from the earth, i.e., near 10 hours from perigee passage. However, unlike the spin-off separation which entails differential velocity increments, the pallet maneuver involves a one-sided velocity increment and the direction of the velocity increment can be chosen so as to avoid decreasing perigee altitude. In this way, a lower injection perigee altitude can be used than would otherwise be acceptable.

An alternate technique for executing an orbital maneuver to obtain larger in-plane normal separation distances is to apply a tangential velocity increment near perigee to reduce the apogee radius and followed by a tangential velocity increment at apogee to increase perigee altitude while restoring the orbital period to its original value. (Alternately, perigee altitude could be increased first.) This maneuver results in a substantial increase in the semi-minor axis and provides in-plane normal separation distances in the section of interest on both legs of the orbit. The velocity increment required at apogee would, of course, be much larger (in the order of 20 times larger) than the velocity increment applied at perigee. For example, to balance a 1.5 meter/sec tangential velocity increment at perigee, which would drop apogee by about 735 km, would require a 29 meter/sec tangential velocity increment at apogee, raising perigee by a similar distance; the semi-minor axis would then be increased by about 1530 km providing in-plane-normal separation distances of this order of magnitude.

In either case the separation distances obtained from the spin-off separations would be at least partially additive to separation distances obtained from the pallet's orbital maneuver. Also, the pallet's orbital maneuver would remove the restriction on the magnitude of the in-plane normal separation distances that stems from the limitations on the velocity increment obtainable from spin-off.

Out-of-plane separation distance could be obtained by firing a solid rocket that thrusts in the axial direction. An advantage that might be gained by using the pallet rather than the satellites to obtain the out-of-plane separation distance is the reduction of accuracy requirements. Previous analysis has shown that accuracy in the direction of firing the axial solid rockets is likely to be one of the most critical factors affecting the growth of tangential separation distance. When the axial rocket is fired from the pallet, any error that is made could be subsequently corrected by observing the pallet's orbital period, and applying pulsed lateral thrusts in the tangential direction to restore the orbital period to its desired value, before separating the remaining satellites from the pallet. In this way the accuracy of the deployment would depend not so much on aspect sensor accuracy but upon tracking accuracies, the delicacy of execution, and the time allowed for the orbital period readjustment. (This comment applies not only to the out-of-plane maneuver but to any pallet orbital maneuver.)

Since the distances between the satellites separated by spin-off are not affected by the pallet's orbital maneuver, it is not possible to obtain large separation distances between these satellites on both sides of the orbit, unless (1) they are separated out-of-plane by firing axial rockets, or (2) the spin-off is deliberately biased to obtain a tangential velocity increment producing a growth of tangential separation distance. Since the former alternative appears to be preferable, it seems unlikely that the use of Deployment Scheme IV would result in complete deletion of the use of satellite axial solid rockets to obtain out-of-plane separation distances.

3.8 DEPLOYMENT SCHEME V - SPIN-OFF SEPARATION OF TWO SATELLITES; AXIAL SEPARATION OF THE OTHER TWO

With Deployment Scheme V there is a choice of many mounting arrangements and separation sequences but none of the choices seem to be of particular merit relative to other deployment schemes. Some of the conceivable mounting arrangements are illustrated in Figure 9. In the first arrangement (Figure 9, Side View A), the two axially-mounted satellites (2 and 4), are connected in line. Thus, during the booster flight inertial loads are delivered from the upper satellite through the lower satellite to the pallet. In this case, the lower satellite must be designed to withstand these loads, and if the satellites are to be identical, the consequent penalty is incured for all of the satellites. The column-supports used for the two laterally-mounted satellites are not ideal for spin-off separation.

In the second arrangement (Figure 9, Side View B), a separate support structure is used to by-pass the loads from the upper axially-mounted satellite around the lower axially-mounted satellite. Both of the axially-mounted satellites are separated in the forward direction. Hence, the separation of the lower axially-mounted satellite will entail a difficult clearance problem unless the surrounding by-pass structure is folded away or separated before the lower satellite is separated. The laterally-mounted satellites could be cantilevered from the by-pass structure, as shown in the figure, or supported by separate columns. In either case, they would be subject to a different load distribution from the axially-mounted satellites so that a single satellite design capable of withstanding either load distribution will entail some penalties.

The third arrangement (Figure 9, Side View C), is similar to the second except that the lower axial satellite is separated to the rear. This is made possible by allowing a bottom port in the pallet to permit passage of the satellite. In this way it may be possible to deal with the clearance problem without folding or separating the by-pass structure.

For arrangements wherein the station of the laterally-mounted satellites conflicts with the station of one of the axially-mounted satellites or with the by-pass structure, as depicted in Figure 9, Side Views B and C, the diameter of the satellites would be more severely restricted than for other alternatives. This disadvantage could be remedied by locating the laterally-mounted satellites at a non-conflicting station, as depicted in Figure 9, Side View A, but this approach requires longer columns.

Clearly none of the foregoing alternatives is particularly attractive from a structural viewpoint. In addition, all of the arrangements present some problem with respect to the attainment of a favorable moment-of-inertia ratio for the pallet.

The need to maintain a favorable moment-of-inertia ratio for the pallet militates against the separation of the laterally-mounted satellites as the first step of the separation sequence. This would leave the separation of one of the axially-mounted satellites as the only choice for the first step, followed either by: (1) separation of the laterally-mounted pair and then the remaining axially-mounted satellite, or by (2) separation of the second axially-mounted satellite and then the laterally-mounted pair. The choice of mounting arrangement must, of course, be compatible with the choice made for the deployment sequence.

It is noted that, although Deployment Scheme V affords two opportunities for executing pallet orbital maneuvers, no advantage is gained relative to Deployment Scheme IV which affords only one such opportunity. The separation distance between the two laterally-mounted satellites will stem only from the spin-off velocity increments unless additional velocity increments (such as the firing of satellite axial rockets) are applied. The situation in regard to the use of the pallet orbital maneuvers instead of satellite axial rockets to obtain out-of-plane separation distances differs in no essential way. Similarly, considerations relative to the use of the pallet orbital maneuvers to obtain larger in-plane normal separation distances are not essentially different. The only major difference is that for Deployment Scheme V the use of the pallet's orbital maneuvers is essential to obtain large in-plane normal separation distances on both legs of the orbit. Hence, while consideration might be given to the use of Scheme IV, with or without pallet orbital maeuvers, there would be little point in considering the use of Scheme V without such maneuvers. Relative to Deployment Scheme IV, the only potential advantage offered by Deployment Scheme V is that only one spin-off separation is required. This could be of some advantage to the extent that the spin-off separations may entail a reliability or accuracy penalty. However, it appears very unlikely that this consideration would offset the major disadvantages of Deployment Scheme V.

3.9 DEPLOYMENT SCHEME VI - AXIAL SEPARATION OF ALL FOUR SATELLITES

In some respects the mounting arrangement problems of Deployment Scheme V are aggravated in the case of Deployment Scheme VI. Thus, if a load by-pass structure is used, as depicted in Figure 10, Side View A, it must provide for by-passing the loads delivered from three satellites instead of just one. Similarly, if the satellites are mounted in a column, as depicted in Figure 10, Side View B, and the loads are transmitted through the satellites, the bottom satellite must support the loads delivered from the upper three satellites. On the other hand, in the absence of laterally-mounted satellites, the diameter of the satellites can be considerably increased and the height of the satellites reduced. Such a squat satellite design would aid the attainment of a favorable moment-of-inertia ratio and assist in the solution of the mounting problems.

In contrast to Deployment Scheme V, in which the reward for the penalty paid in handling the mounting problem is meager at best, Deployment Scheme VI offers a potentially large reward. It permits in-plane normal separation distances, larger than could be obtained by spin-off separation, to be obtained on both legs of the orbit; it permits and, in fact, can assure the attainment of a markedly non-coplanar array, without the need for growth of tangential separation distances; and, most important, it can provide maximum assurance of the avoidance of excessive growth of tangential separation distances without requiring the use of on-board systems of very high precision. The latter feature is obtained by avoiding the application of large velocity increments to the satellites either during or after separation. Since the orbital period of the pallet can be measured very precisely by ground tracking and can be corrected prior to each satellite separation, the residual error can be made extremely small. Since the separation velocity increments applied to the satellites will be quite small, and will nominally be in the direction normal to the orbit, the effect of the separation velocity increments on orbital period should be negligible, particularly when the separations occur at large distances from the earth where the sensitivity of orbital period to velocity increments is small.

The order in which the tangential, out-of-plane and in-plane-normal separation distances are generated can be varied from that given in Figure 10. Factors to be considered in selecting the order are: (1) impulse requirements, (2) time required to complete the deployment, and (3) influence on the array of separation distances. Of these factors the former is likely to be of most importance. The order given in Figure 10 was predicated on this consideration. Since the tangential separation distance requires the smallest velocity increment, impulse requirements are likely to be reduced by obtaining this component first. Since the out-of-plane velocity increment can be generated by firing an axial solid rocket, whereas, if repeated attitude reorientation maneuvers are to be avoided, the in-plane-normal velocity increment will require the use of pulsed lateral thrust, the out-of-plane velocity increment should be more readily generated and, therefore, was given precedence.

The technique for obtaining the various separation distance components has been described above in the comments relating to Deployment Scheme IV. For Deployment Scheme VI as presently contemplated, the out-of-plane velocity increment would be applied near apogee to obtain large out-of-plane separation distances on both sides of the orbit. To obtain large in-plane normal separation distances on both sides of the orbit, the technique of raising perigee altitude and reducing apogee radius would be followed.

In addition to obviating the need for very precise aspect data and orientation of the pallet's spin axis, the complete absence of spin-off separation deletes the problems associated with this type of separation and the costs in time and money for the development test program required to resolve these problems.

3.10 EVALUATION LOGIC

Table 4 summarizes the comparison of the deployment schemes. Inasmuch as Deployment Scheme III appears to offer no possibility of providing any advantage over Scheme I it was not included in the summary. Since Scheme IV conceivably could be applied either with (IV-A) or without (IV-B) an orbital maneuvering pallet, these options were tabulated separately.

Basically, there are two ways to improve the performance of Deployment Scheme I: (1) the use of additional spin-off separations, and (2) the use of pallet orbital maneuvers. The former device by itself can provide mainly for large in-plane-normal separation distances on both legs of the orbit. However, its potential for increasing the maximum magnitude of the in-plane-normal separation distance is limited and it does not avoid the need for growth of tangential separation distance to obtain non-coplanarity. The latter device can by itself remedy all of the Scheme I shortcomings. It does, however, involve the use of more complex pallet operations.

Since the developments associated with spin-off separation and the design of an orbital maneuvering pallet are both major items, and it does not appear possible to avoid both, it would be desirable to emphasize deployment schemes that, at least, avoid one or the other. Thus, consideration

Table 4

DEPLOYMENT SCHEME COMPARISON SUMMARY

	I	II	IV-A Without Pallet Orbital Maneuver	IV-B With Pallet Orbital Maneuver	V	VI
1. Number of Spin-Off Separations	1	3	2	2	1	None
2. In-Plane Normal Separation Distance						
Large separation distance attainable on both legs of orbit?	no	yes	yes	yes	yes	yes
Magnitude limit imposed by spin-off velocity increment	~800 km on one leg; on other leg, separation distance very small	~1200 km on one leg ~800 km on other	~800 km on both legs	not limited	not limited	not limited
3. Tangential Separation Growth Required for Array Non-Coplanarity?	yes	yes	yes	no	no	no
4. Avoidance of Excessive Growth of Tangential Separation	Difficult	Most difficult	More difficult	Somewhat easier if satellite solid rockets are not used	Somewhat easier if satellite solid rockets are not used	Easiest
5. Perigee Altitude Required at Injection, km	750	810	750	750	750	630
6. Attainment of Favorable Moment of Inertia Ratio	Difficult for pallet; impracticable for satellite pairs	Easy for all configurations	Easy for all configurations	Easy for all configurations	Most difficult for pallet	Some difficulty for initial pallet configuration
7. Structural Design Considerations	Fair	Good	Good	Good	Bad	Poor
8. Ease of Boom Stowage	Fair	Fair	Good	Good	Poor	Poor
9. Complexity of Pallet and Deployment Process	Minimum complexity	Relatively simple pallet	Near minimum complexity	Complex	Complex	Most complex

might be divided between schemes that do not entail an orbital maneuvering pallet, and schemes that do not entail spin-off separation. In the former category it appears likely that Scheme IV-A could offer sufficient improvement over Scheme I to justify the additional complication of a second spin-off, while the improvement relative to Scheme IV-A obtained with Scheme II may not warrant the complication of a third spin-off.

The only scheme that avoids spin-off separation is Scheme VI. Moreover, of the schemes that make use of an orbital maneuvering pallet, only Scheme VI reaps the full benefits of the use of this device. Thus, Scheme VI was selected as the most promising alternate for detailed analysis and comparison with the original multiple satellite reference configuration, since it best meets the improvement criteria which were established for the alternate system study.

Section 4

SCHEME VI ANALYSIS

Deployment Scheme VI, as illustrated in the previous section, has advantages which warrant further consideration. Analysis has been performed to optimize the deployment operation with respect to propulsion requirements, time to complete deployment, separation distances of the array and the influence of perturbations. The sequence of the maneuvering, their sense, and the total time to complete the deployment have been established.

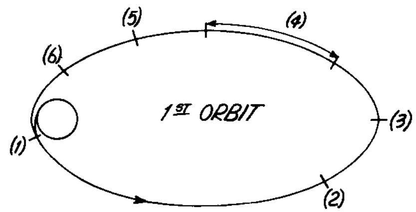
The design feasibility of the Deployment Scheme VI concept has been investigated and feasibility established. The design requirements were determined and configuration studies performed. In conjunction with the configuration studies, analyses were conducted in the areas of structural design, lateral thrusting system/attitude control system, antenna design, power system design, and thermal control. A recommended concept has evolved and weight statements for satellites and the total system are presented.

The analytical and design efforts performed for Deployment Scheme VI are summarized in the following sections.

4.1 SCHEME VI OPTIMIZATION

The operational deployment sequence, illustrated in Figure 12, is based upon the results of an optimization study. The deployment was optimized with respect to: (1) propulsion requirements, (2) time required to complete the deployment, and (3) array characteristics obtained. The main factors that affect these characteristics are the order in which the orbital maneuvers are made and the sense of the maneuvers. The selected order

1st Maneuver:	Tangential
2nd Maneuver:	Out-Of-Plane
3rd Maneuver:	In-Plane Normal



1ST ORBIT

(1) SEPARATE 3RD STAGE, PALLET BEACON ON
UNCAGE PRECESSION DAMPERS,
SATELLITE XMTRS ON (ASPECT DATA)

(2) DEPLOY BOOMS

(3) ΔV TO INCREASE PERIGEE (SOLID ROCKET)

(4) ACS + TO ORBIT PLANE

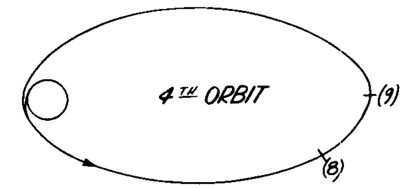
(5) SEPARATE SATELLITE #1

(6) 1ST ΔV_T OF TANGENTIAL MANEUVER - CHANGES ORBIT PERIOD

2ND ORBIT - LEAD TIME BUILDS UP

(7) 2ND ΔV_T OF TANGENTIAL MANEUVER APPLIED NEAR PERIGEE - RESTORES ORBIT PERIOD

3RD ORBIT - TRACK, TRIM ATTITUDE AND ORBITAL PERIOD

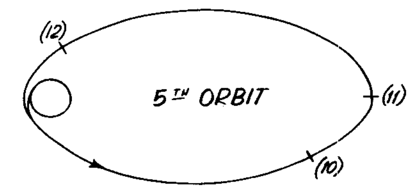


4TH ORBIT

(8) SEPARATE SATELLITE # 2

(9) Δ V OUT-OF-PLANE (SOLID ROCKET)

TRACK AND TRIM



5TH ORBIT

(10) SEPARATE SATELLITE #3

(11) APPLY FIRST Δ V OF IN-PLANE NORMAL MANEUVER
3RD ΔV_T - INCREASES PERIGEE

(12) APPLY SECOND Δ V OF IN-PLANE NORMAL MANEUVER
- LOWERS APOGEE AND RESTORE PERIOD

6TH & 7TH ORBITS - TRACK, TRIM ATTITUDE AND ORBITAL PERIOD OF SATELLITE # 4

Figure 12. Operational Sequence

minimizes propulsion requirements and is at least equal to the alternatives in other respects. The propulsion requirements are defined in Section 4.2 and are based upon the attainment of intersatellite separation distances of 1500 km in each of the orthogonal directions on both legs of the orbit in the region of most experimental interest. The deployment operations nominally require 6-1/2 orbits, or about 13 days.

The optimum selection of the sense of the maneuvers was found to be as follows:

- The sense of the period change used in the tangential separation is selected so that the first satellite separated, Satellite 1, lags the others.
- The velocity increment used to generate the out-of-plane separation distance is applied in the downward direction, i.e., in the direction opposite to the positive normal to the orbit.
- The sense of the in-plane normal maneuver is chosen so that the perigee altitude change produced by the maneuver is positive.

The advantage of increasing instead of decreasing perigee altitude is obvious. Other advantages of this selection of the sense of the maneuvers are:

- The adverse effects of subsequent orbital perturbations on the array characteristics are minimized. In particular, the effects on the in-plane normal separation distances obtained in the section of the orbit of most interest (near 12 R_e) are reduced to a growth at a rate of about 100 km/month on the ascending leg and a decrease at a similar rate on the descending leg.
- In the nominal case, the out-of-plane maneuver will produce a reduction in the inclination of the orbit relative to the ecliptic plane.
- The effects of inadvertent growth of the tangential separation distances on the non-coplanarity of the array is minimized.

By maintaining precise control of the satellite's orbital periods, excessive growth of tangential separation distances is avoided. The following features are incorporated in the operational sequence to optimize the control of orbital period:

- Except for the first satellite, the separation of satellites is executed at large distances from the earth. At these distances the sensitivity of orbital period to the separation velocity increment is very small.

- Several opportunities for measuring and trimming orbital period are provided prior to the separation of the satellites. This assures convergence of the cut-and-try process used to control orbital period.

The in-plane normal separation distances are obtained by means of a two-step maneuver. The first step is the application of a tangential velocity increment near apogee. This increases the orbit's perigee altitude and also increases the orbital period.

The second step is the application of a tangential velocity increment close to earth (ideally at perigee). The second velocity increment reduces apogee altitude and nominally restores the orbital period to its original value. In-plane normal separation distances in the section of the orbit of main interest are obtained as a consequence of the "fattening" of the orbit, i.e., the increase in the minor axis produced by the maneuver.

Because of its capability of applying multiple velocity increments and correcting orbital period changes, Scheme VI opens a wide range of possible variations of the in-plane normal maneuver. A study was made to explore these possibilities. The results obtained indicated that:

- The large change in period that is encountered during the first step of the maneuver cannot be avoided without a substantial increase in propulsion system weight ($\sim$ 5 pounds).

- Shifting of the point of application of the second velocity increment from perigee to about 3 R_e on the descending leg to avoid solar occultation and to provide better position and visibility for tracking purposes results in a small increase in propulsion requirements. (This increase has been taken into account.)

- A slightly asymmetrical distribution of in-plane normal separation distances (e.g., 1,800 km on the descending leg; 1,200 km on the ascending leg), which is desirable in view of the effects of the subsequent orbital perturbations and the sunline orientation, can be obtained without a significant increase in propulsion requirements.

Consideration of the optimum time for deployment of booms indicated:

- The booms should be deployed prior to the separation of any satellite to simplify the problem of attaining the desired spin rate of 60 rpm for the satellites and for the pallet/satellite combination. (See Section 4.2.2 and 4.2.4 for discussion of satellite and pallet configurations and satellite/pallet integration.)

- It appears preferable to deploy the booms prior to the execution of any maneuvers. The reduced spin rate obtained upon deployment of the booms and avoidance of the necessity for operation at two different spin rates should aid the design of the ACS and LTS systems.

4.2 SCHEME VI SYSTEM DESIGN

As described in Section 3.9, Deployment Scheme VI involves a radically different concept than was previously considered in connection with the Scheme I design. In this case, the satellites and pallet are mounted in series and the satellites are separated in the axial direction. The purpose of this section is to examine the design implications of this concept.

4.2.1 DESIGN REQUIREMENTS

A number of requirements on the system configuration for the Scheme VI deployment are derived from the operational sequence and deployment functions. The primary requirements on design, particularly those that differ from the original reference configuration, are presented in Table 5.

4.2.2 CONFIGURATION STUDIES

<u>CONFIGURATION A</u>

Deployment Scheme VI permits the diameter of the satellites to be increased to about 50 inches. This allows the solar cell surface area needed to satisfy power requirements to be obtained with a satellite height of only 9 inches. However, if the collinear array antenna is retained, the antenna must protrude nearly 15 inches beyond the satellite. The spacing between the satellites, when mounted on the pallet, must then be extended to allow

Table 5

SCHEME VI DESIGN REQUIREMENTS

- Balance
  - Favorable F_{Ratio}, All Configurations
  - Accurate C. O. M. Location When Thrusting
  - All Booms Deployed Prior to 1st Satellite Separation

- Propulsion
  - Solid Rockets

    30 M/sec Perigee Increase ~ 1230 lb-sec

    30 M/sec Out-Of-Plane ~ 720 lb-sec

    Thrust Along Spin Axis Through C. O. M.

  - Lateral Thrusting System (LTS)

    4 M/sec Tangential ~ 120 lb-sec

    35 M/sec In-Plane ~ 525 lb-sec

    Pulsed Thrust Laterally
    (Through C. O. M., Each Configuration)

  - Attitude Control (ACS) ~ 525 lb-sec

- Pallet Lifetime - 6 1/2 Orbits, Minimum room for the antennas. This makes the attainment of a favorable moment-of-inertia ratio for the initial assembly very difficult.

A design, Configuration A, using the collinear array antennas, with the satellites stacked above the pallet, is illustrated in Figure 13. The "doughnut" satellite shape allows the antennas to be nested, i.e., the antenna of a lower satellite extends up through the hole of the upper satellite, thus permitting the space between the satellites to be reduced to about 7 inches.

The satellites are joined through their cylindrical "cores," which are about 21 inches in diameter, the satellite-to-satellite and satellite-to-pallet connections being made through a band and "V" block assembly. The 7-inch space between the satellites provides clearance for hook-up and accessibility to the separation systems. Satellite loads are tranmitted to the pallet through the central cylindrical cores. A smaller diameter band and "V" block assembly (approximately 9 inches), that is supplied by the launch vehicle agency as part of the "FW-4D spacecraft attach fitting," is used to join the pallet to the fourth stage motor. Access to the FW-4D igniters, attach bolts and electrical connections is unhampered in accordance with a prime requirement of the launch vehicle agency.

Inertia calculations indicate that for this design, with no booms deployed, the roll moment-of-inertia of the initial assembly will be about 38 slug-ft^2, while the pitch and yaw moments of inertia will be about 63 slug-ft^2, yielding a moment-of-inertia ratio of about 0.6.

This unfavorable ratio cannot be significantly improved by the deployment of the satellite's booms. The deployment of a satellite boom increases the roll moment-of-inertia by about 2 slug-ft^2. If 8 satellite booms (2 per satellite) could be deployed in orthogonal directions at the center of mass station, the spin moment-of-inertia would be increased by 16 slug-ft^2 to a value of 54 slug-ft^2, the pitch/yaw moment-of-inertia would be increased by 8 slug-ft^2 to a value of 71 slug-ft^2. Thus the moment-of-inertia ratio would remain strongly unfavorable.

Due to the axial separation of the satellites, their booms cannot feasibly be deployed from the center of mass station. Off-center boom

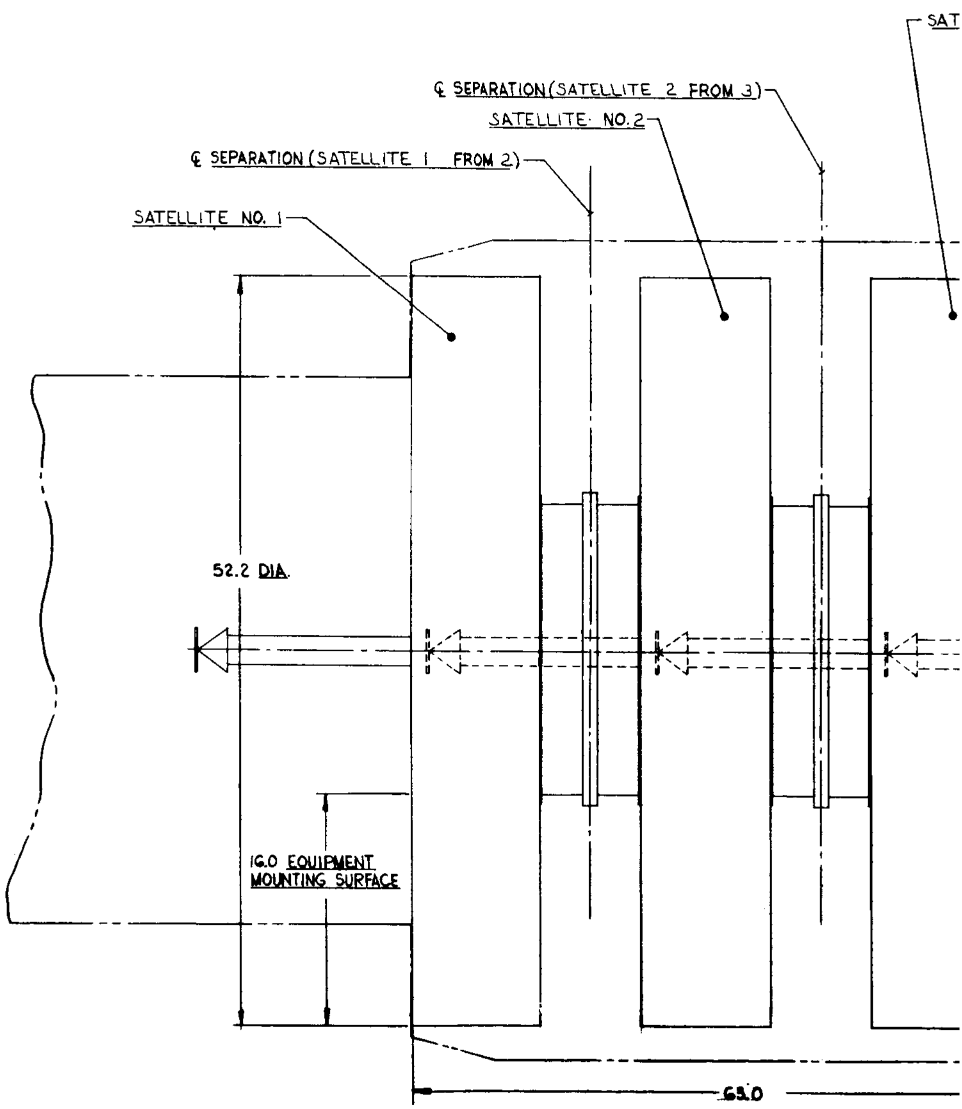
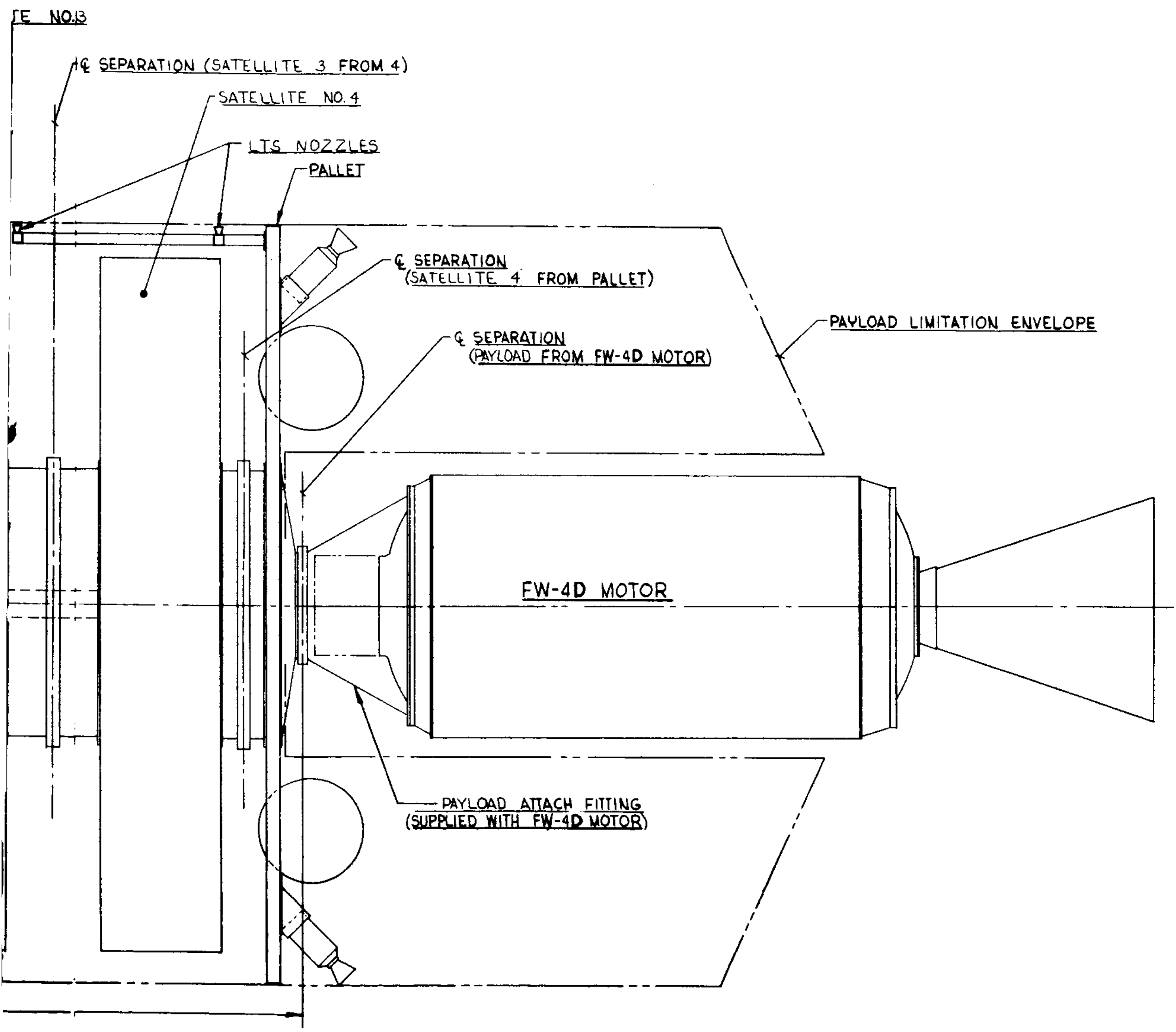


Figure 13. Sche



VI Design Configuration A

deployment would not affect the contribution to the spin moment-of-inertia but would increase the contribution to the pitch/yaw moment-of-inertia. Hence, the deployment of the satellite booms would be even less effective than indicated by the foregoing estimate.

To obtain a favorable moment-of-inertia ratio, additional booms, i.e., booms deployed from the pallet, would be required. At best, the increase in roll moment-of-inertia due to any boom deployment is twice the increase in pitch/yaw moment-of-inertia. On this basis, to attain a moment-of-inertia ratio of 1.1, a minimum increase in roll moment-of-inertia of about 35 slug-ft^2 is required. If the pallet booms are to be of about the same length as the 6-foot satellite booms, then the weight to be deployed must be about 30 pounds. Deploying a weight of this size and/or significantly increasing boom length will undoubtedly present formidable problems in boom design. Furthermore, the need to increase the initial spin rate substantially to offset the greater reduction in spin rate that would be incurred by deployment of the pallet's booms would aggravate the boom design problem. Also, the bottom location of the pallet is undesirable for boom deployment, inasmuch as the effectiveness of the booms is degraded when the deployed weights are not at the CM station.

Additional objections to this design are: (1) because of the nesting of the antennas, the downlink from only one satellite is available when the satellites are still on the pallet; and (2) a relatively high stand is needed for mounting the lateral thrusting system's (LTS) nozzles. The high stand presents a possible obstacle that must be cleared when the last satellite is separated.

CONFIGURATION B

From the results obtained with Configuration A it appears that, for Deployment Scheme VI, the use of a collinear array antenna would be highly undesirable if not impracticable. The cavity-backed, slotted array antenna described in a following section is more suitable since it does not require any protuberances to extend beyond the 9-inch height of the satellites. A design, Configuration B, based on the use of an antenna of this type is shown in Figure 14.

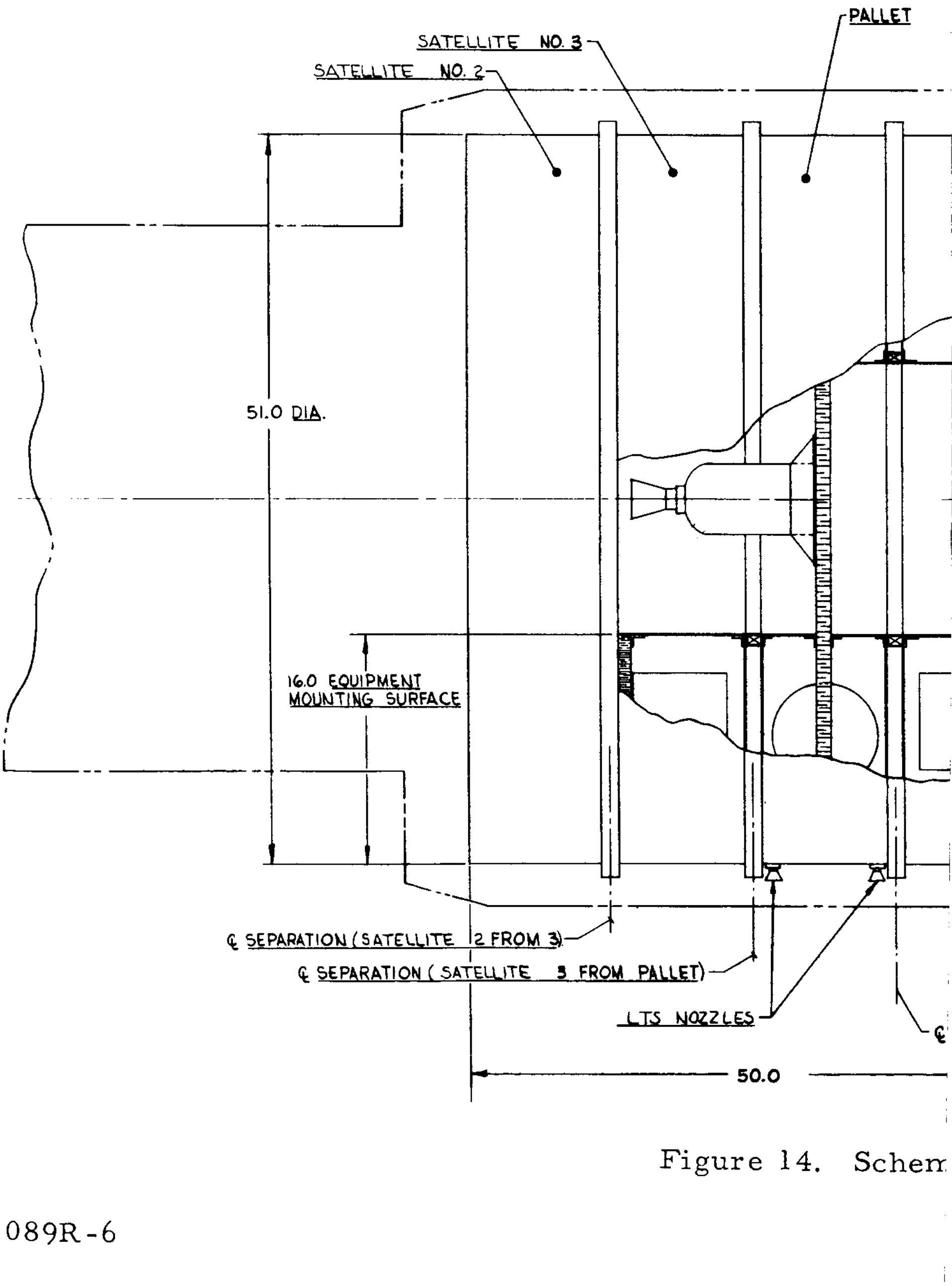
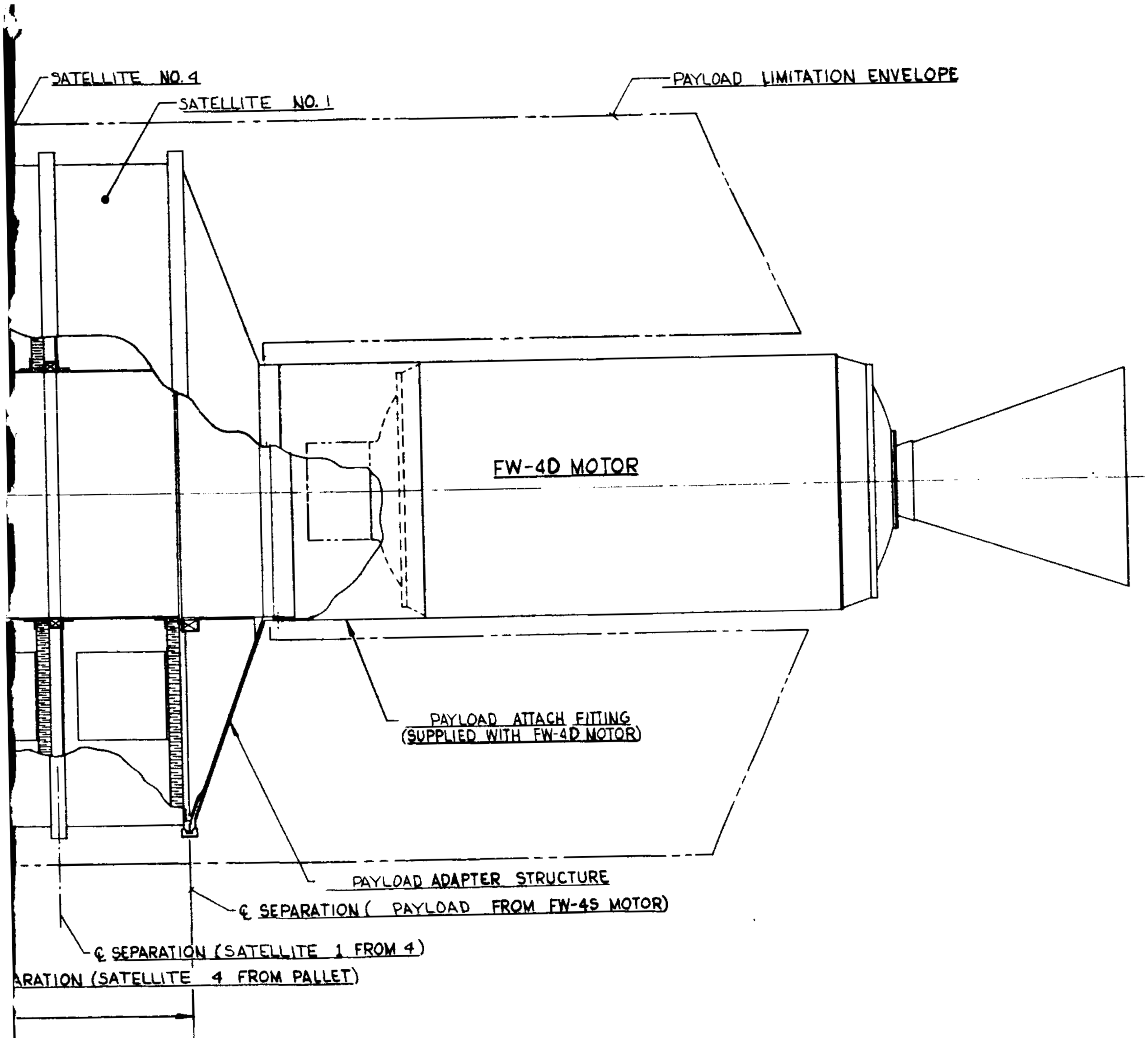


Figure 14. Schem



VI Design Configuration B

In this design the pallet is about 9 inches in height and is centrally located with two satellites above and two satellites below. The central location allows the LTS nozzles to be mounted within the height allowed for the pallet. The outer skins of the satellites and the pallet form the main structural members for transmission of loads through the satellites and the pallet. The satellite-to-satellite and satellite-to-pallet connections are made at the outer diameter by band and "V" block assemblies. A similar connection is made at the bottom satellite to a payload adapter structure which is attached to the FW-4Dpayload attach fitting.

The roll moment-of-inertia of the initial assembly for Configuration B would be about the same as for Configuration A, i.e., about 38 slug-ft^2; it is estimated that the pitch/yaw moment-of-inertia would be reduced to about 36 slug-ft^2, yielding a slightly favorable moment-of-inertia ratio of about 1.06. More definitive evaluation of component weights would be required to ascertain whether an acceptable minimum ratio of 1.1 can be obtained. However, there is no doubt that a satisfactory moment-of-inertia ratio would be obtained if all of the satellite booms are deployed while the satellites are mounted on the pallet.

In this case, it would be desirable to deploy three booms from each satellite since, with fewer booms, the attainment of the desired 60 rpm spin rate would be awkward. With an initial spin rate of about 100 rpm the deployment of 3 booms would drop the spin rate to the desired 60 rpm. The use of three booms also retains a symmetrical inertial distribution at all stages of the separation sequence and would provide a favorable moment-of-inertia ratio of about 1.3 for the initial assembly. With fewer booms the initial spin rate for the booster would be unacceptably low. Because of the close spacing of the satellites, telescoped booms that are stowed within the satellite envelope seem likely to be a most suitable choice.

CONFIGURATION 2 (C)

Since the volume available in a cylinder 9 inches high and 50 inches in diameter is far more than is required for housing all of the pallet's equipment, plus all the equipment of one satellite, it is possible to improve the Configuration B design by incorporating the pallet and one satellite within a 9 inch or, perhaps, slightly greater height. This can be most readily accomplished if the last satellite is not to be separated from the pallet. Either an integrated pallet/satellite unit can then be used, or the satellite and pallet housings can be separate, with all of the satellite housings identical but designed so that any satellite can be mated with the pallet to form a pallet/satellite unit of about the same height as the separate satellites.

A design, Configuration C, of this type is shown in Figure 15. In this design Satellite 1 is mounted below the pallet/satellite combination; while Satellites 2 and 3 are stacked above the pallet/satellite combination. The connections of satellite-to-satellite and satellite-to-pallet/satellite are made through ball-lock separation mechanisms. These mechanisms are mounted at the outer diameter, being recessed just enough to avoid solar cell shadowing. Since both Satellite 1 and Satellite 3 are directly attached to the pallet/satellite combination, only Satellite 2 attaches to another satellite and only Satellite 3 transmits loads from another satellite (i.e., Satellite 2). The pallet/satellite combination is attached to the FW-4D payload attach fitting through an adaptor structure which raises the payload assembly enough to provide clearance below the bottom satellite for access to the FW-4D igniters, attach bolts and electrical connections.

The pallet's solid rockets and the LTS cold-gas tanks are mounted at the hollow of the satellites. Liners around the satellites' inner diameter protect the satellites from the rocket plumes.

Insulation material at the interstices between the satellites is provided to maintain the efficiency of the slotted array antennas when the antennas are used for downlink transmission while the satellites are still attached to the assembly. This material also provides protection against bumping of the satellites when they are separated.

The LTS nozzles are mounted at the outer diameter of the pallet/satellite combination and are recessed to avoid shadowing. The ACS nozzle extends somewhat beyond the outer diameter and overhangs the bottom satellite slightly.

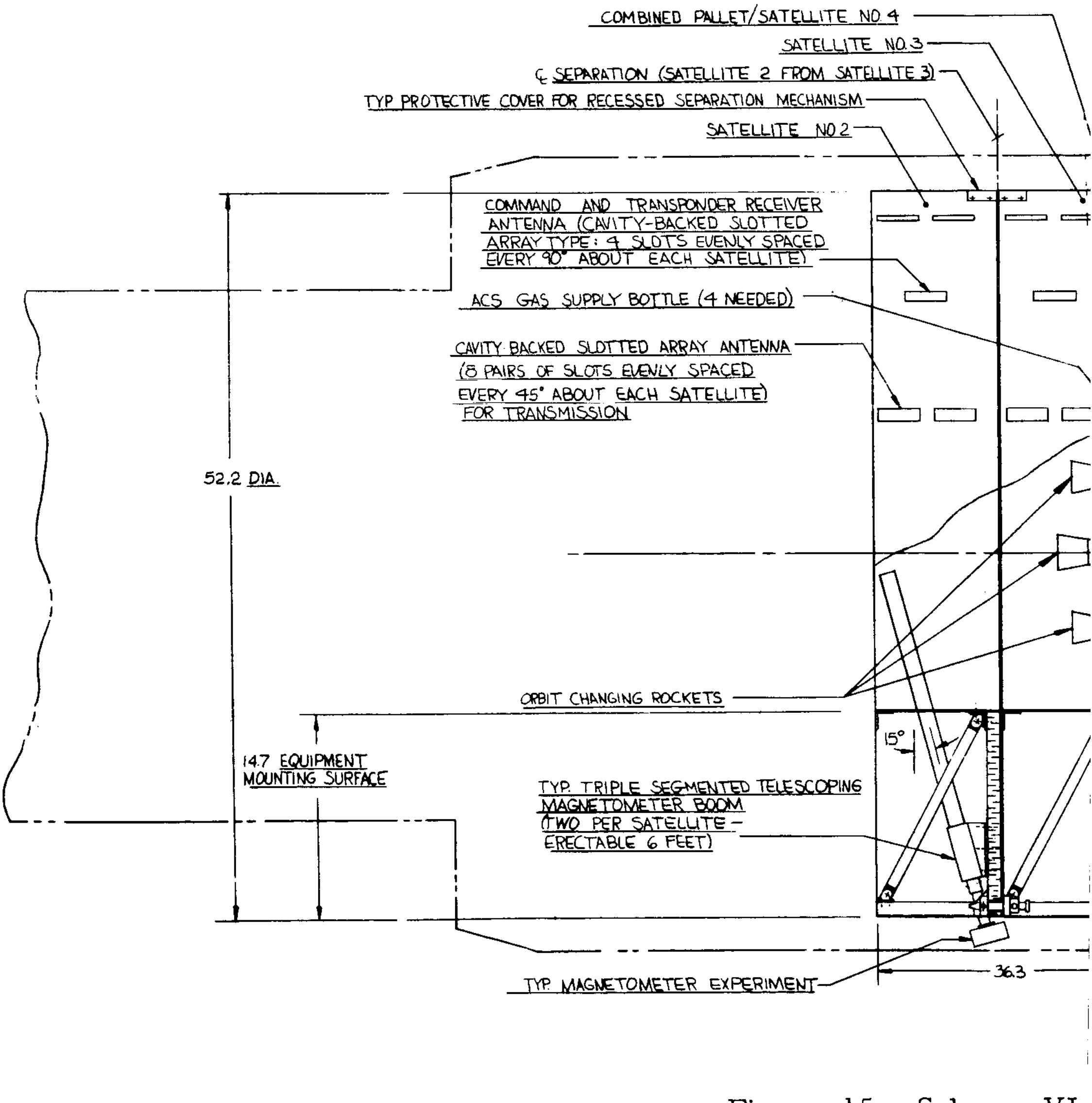
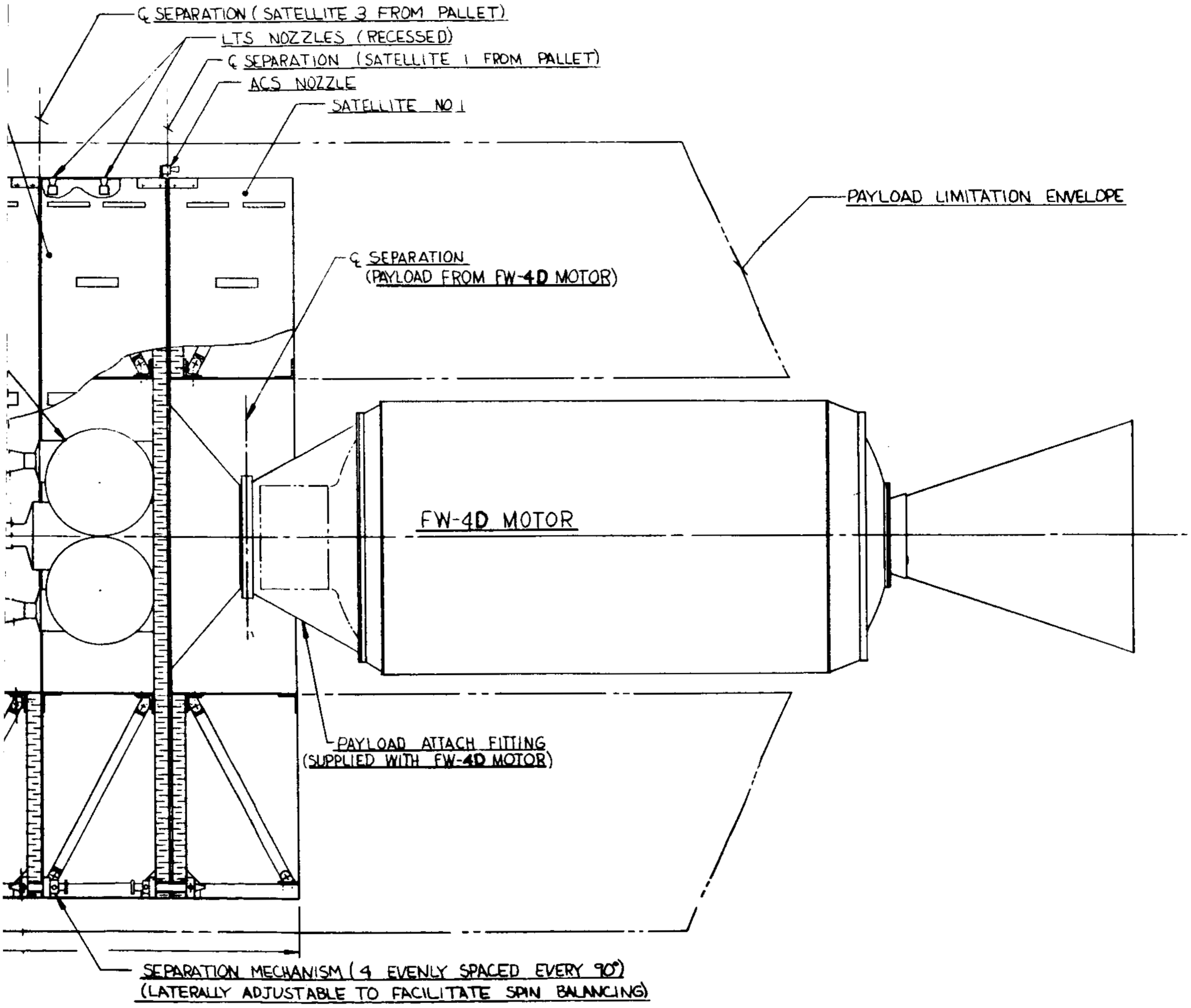


Figure 15. Scheme VI



Design Configuration C

Because of the mounting of most the pallet weights close to the longitudinal axis, the roll moment-of-inertia of the payload assembly will be smaller than for the other configurations. It is estimated that the payload's roll moment-of-inertia will be about 31 slug-ft^2. After separation and deployment of the two satellite booms, the roll moment-of-inertia will increase to about 49 slug-ft^2. This increase would reduce an initial spin rate of 95 rpm to the desired 60 rpm.

The reduction of payload height and the more central location of the pallet's weight results in a substantial improvement in moment-of-inertia ratio. It is estimated that the ratio for the initial assembly will be about 1.34 with booms stowed and will increase to about 1.42 when the booms are deployed. With the successive separation of satellites, the moment-of-inertia ratio of the remaining assembly will generally increase. The separated satellites, which are assumed to have only two booms that are 180 degrees apart, will still have a moment-of-inertia ratio greater than about 1.4.

Configuration C appears to provide a reasonably compact and efficient design that is at least competitive with the Scheme I design. While some problems may be uncovered upon further study, no serious obstacle to the development of this design is apparent.

4.2.3 STRUCTURAL DESIGN

The selected satellite structural design (Configuration C) consists of a single, annular all-aluminum honeycomb transverse plate which supports the instruments and subsystems. The loads are transmitted through vertical stringers around the periphery of the satellite and around the internal annulus (not shown in Figure 15). Each satellite is internally supported with diagonal struts which transfer loads to the stringers. The outer shell of the satellite, except for the bottom plate, carries no load. The top plate is used primarily for thermal control, the external shell for support of the solar cells, and the internal shell of the annulus for insulation against the rocket exhaust.

The pallet/satellite combination is very similar in design to the satellites, except that the inner void of the satellite annulus is used for mounting the pallet components. The transverse plate is a disc instead of annular in shape. A conical adapter attached to the disc plate is used to mount the pallet on the booster vehicle.

4.2.4 PALLET/SATELLITE INTEGRATION

Considerations have indicated that for Scheme VI it would be possible to forego separation of the last satellite from the pallet and that it would be advantageous to use a more or less integrated pallet/satellite design which combines the last satellite with the pallet. The main advantages that could be gained by this approach are:

- A reduction in the height of the initial assembly of the pallet, plus the satellites, can be effected. This assures the attainment of a favorable moment-of-inertia ratio and reduces booster flight loads.

- A saving in weight can be obtained through:

  Deletion of one separation system

  Reduced structural weight for the pallet/satellite combination

  The sharing of subsystems between pallet and satellite, e.g., power, command receiver, etc.

  Carrying experiments that require only single-point data on the pallet/satellite combination only.

  Avoidance of the need for a separate precession damper for the pallet.

- With the use of a pallet/satellite combination, a solar power source can be made available for the deployment operation. With Scheme VI, the deployment operation is more complex and the total deployment time is subject to greater variation. If only battery power were available, a hard limit would have to be imposed upon the allowable deployment time and the deployment operation would then be less secure.

- With Scheme VI the availability of a downlink will be extremely helpful if not mandatory. In order that the most intelligent decisions be made regarding the use of the cold-gas supply common to the attitude reorientation and lateral thrusting systems, data on the remaining supply must be continually available. Separate pallet downlink telemetry would entail a substantial increase in weight and cost.

- To the extent that some cold gas will remain after completion of the initial deployment, some latitude is available for readjustment of the orbit of one of the satellites. This latitude provides a degree of adaptability which could be used to compensate for differential orbital perturbation effects that may distort the array during the latter portion of the operational lifetime, and/or for generating changes in inter-satellite separation distances which may be indicated, by the scientific data obtained, to be advantageous for the purposes of the experiment.

The main objections to the use of a satellite/pallet combination are (1) loss of commonality between all of the satellites, and (2) the need for the pallet design to maintain the same degree of magnetic cleanliness required for the satellites. The latter objection is not considered to be a serious obstacle since to avoid any possibility of contaminating the satellite it had been intended that a maximum degree of magnetic cleanliness would be maintained for the pallet even if all satellites were to be separated from the pallet. Also, while the pallet's systems will be more complicated, no new types of elements are required that were not present on Pioneer 6. Pioneer 6's attainment of a satisfactory degree of cleanliness would, therefore, indicate that meeting the magnetic cleanliness requirements for the pallet will not be unduly difficult.

With regard to commonality, it is noted that, even if a totally integrated pallet/satellite combination were used, the number of distinct uninterchangeable units would not be increased. With a pallet that is entirely separate the total system would be comprised of four identical, interchangeable satellites and one pallet; with an integrated pallet/satellite combination the total system is comprised of three identical, interchangeable satellites and one pallet/satellite combination.

It is further noted that a considerable degree of commonality could be retained depending upon the philsophy that is followed in the design of the pallet/satellite combination. At one extreme, commonality would be totally lost if an approach were followed that called for the maximizing of the direct benefits, mainly weight reduction, that could be gained through a totally integrated design. In this case, the pallet/satellite combination would be an entirely separate and distinct unit. At the other extreme, practically no loss in commonality would be incurred if an approach were followed that required the satellite part of the pallet/satellite combination to be identical to other satellites with the only integration provision being that it be structurally compatible with the pallet to form an assembly of suitably restricted envelope. This approach would simplify the design process because the design of integrated subsystems would be avoided, but many of the potential benefits of an integrated pallet/satellite design would be lost.

The most serious objection to following the latter minimum-integration, maximum-commonality approach is, simply, that a more reasonable degree of integration is necessary to avoid an excessive weight penalty. However, it appears that a satisfactory compromise can be pursued that does not impose an excessive weight penalty, yet retains the main advantages of an integrated design as well as a considerable degree of commonality. This approach calls for the assembly of the pallet/satellite combination from: (1) a satellite that is identical to the other satellites, and (2) unique parts required for the pallet/satellite combination. In the assembly process, parts of the satellites may be removed and replaced by the unique items required for the pallet/satellite combination. In this concept any satellite could be selected for the assembly of a pallet/satellite combination. However, the assembly would be a factory process. A separate satellite would not be interchangeable with the satellite of the pallet/satellite combination in the field. Hence, a separate pallet/satellite combination spare would be required to back up launch operations.

It is noted that with the main pallet elements housed in the hollow of the doughnut-shaped satellites some modification of the satellite is practically unavoidable during the assembly of the pallet/satellite combination, in order to provide exterior access for the ACS and LTS nozzles. To provide exterior access for these as well as other pallet elements, such as the beacon antenna and the digital solar aspect sensor, satellite modifications would be required to minimize the height of the pallet/satellite combination.

The compromise integration concept implies that provisions for mating with the pallet will be included in every satellite assembly. For example, if the design of the pallet/satellite combination calls for electrical interconnections between a satellite and a pallet subsystem, all of the satellite systems would be wired with the appropriate junction boxes and/or connectors.

The repetition on each satellite of such elements, that are utilized only by the pallet/satellite combination, will entail some increase in weight. However, it is felt that this weight penalty can be kept small enough to justify the simplification of design and fabrication that is gained.

The compromise approach allows considerable flexibility with respect to the degree of integration or commonality that is retained. It permits a separate decision to be made for each of the subsystems where sharing of pallet and satellite functions is possible. An optimum selection can be made on the basis of total system considerations. In subsequent sections the main subsystems involved in the pallet/satellite integration are briefly discussed.

4.2.4.1 ELECTRICAL POWER SYSTEM

If all pallet systems were to be separate from the satellite systems, the minimum power requirement would be as follows:

Command Receiver/Decoder	468 watt-hours
Beacon	3
ACS/LTS	5

To satisfy this requirement, using a separate pallet battery supply, would require about 10 pounds of batteries and an additional 1.5 to 2 pounds for power conditioning and cabling. (The provision of a separate pallet telemetry downlink would considerably increase electrical power requirements.)

The required pallet battery weight could be reduced to about 5 pounds, while retaining near maximum commonality for the satellite subsystems by using a trickle charge from the satellite's solar power system. However, for a more integrated design wherein a single command receiver/decoder serves the pallet as well as the satellite systems, the additional power required for the separate pallet functions would be reduced to about 8 watt-hours. This additional power requirement could be easily accommodated by the satellite's power. The satellite's power supply could accommodate the pallet's requirements even if a separate pallet command receiver/decoder were used, since during deployment the power demand of the satellite's subsystems would be far below that which is required after the satellites are deployed and are fully operational.

It is estimated that the modification of the satellite's power distribution system that would be required in order for this system to supply power for the pallet's functions will not be in excess of .5 pounds.

4.2.4.2 COMMAND/RECEIVER DECODER

A separate pallet command receiver/decoder would weigh about 3.0 pounds. In addition to this weight penalty the use of a separate pallet command/receiver decoder hardly merits consideration because of the impracticability of providing a separate antenna for the pallet's receiver. If the receiver antenna is to be shared there would be little point in avoiding a common receiver/decoder. It is estimated that the satellite's receiver/decoder, which would weigh about 3.0 pounds, could be replaced by a unit that would weigh about 3.5 pounds and would be capable of handling all satellite and pallet functions. Thus, for an integrated design, the pallet's command functions would be satisfied at an incremental cost of only about .5 pounds and the complications of a separate antenna system would be avoided. In addition, the use of the integrated system will improve reliability.

4.2.4.3 THERMAL CONTROL

Preliminary investigation indicates that the degree of integration will have little effect on the analysis, design and development work required in connection with thermal design. Invariably, it will be necessary to give consideration to: (1) the individual separated satellites; (2) the separated pallet/satellite combination, and (3) the assemblies that appear during the deployment process. However, if all the satellites are identical, there is no freedom to design the pallet/satellite combination differently, which restricts the design and may produce differing thermal control characteristics in the pallet/satellite combination than in the individual satellites.

4.2.4.4 TELEMETRY DOWNLINK

Not including the costs of a separate power supply, which would be prohibitive, or the difficulty of providing a separate antenna system, a separate pallet real-time-only data collection and transmission system would entail a weight penalty of about 6.5 pounds. With the use of the satellite's downlink, telemetry from the pallet can be obtained by a minor modification of the satellite's data processor.

4.2.4.5 STRUCTURE

The main question with regard to the structure of the pallet/satellite combination is the extent to which the satellite's structural members are to be dismounted and replaced by integral members. The configuration drawing shown in Figure 15 shows an integral honeycomb disc-shaped panel which replaces the satellite's annulus panel. Instead of an integral disc, the satellite's annulus panel could be retained. A separate disc panel could then be attached to the satellite's panel with bolts to complete the connection through the adaptor structure to the booster vehicle's payload attachment fitting. With the integral panel design, it is estimated that the structure of the pallet/satellite combination would be only about five pounds heavier than the separate satellite's structure. Retention of the satellite's annulus panel would increase the weight of the pallet/satellite structure by about 2.5 pounds.

It should be recognized that insufficient design work has been done in connection with Scheme VI to explore all the possibilities with respect to the structural design of the satellites and the pallet/satellite combination. Although the design illustrated in the above-referenced figure appears to be acceptable and competitive with the design contemplated for Scheme I, it is possible that further study will disclose an even superior alternative. It is noted that, in evaluating the structural design, more consideration must be given to the total weight of the combined structure, including all of the satellites, than to the increment of the weight of the pallet/satellite combination over the weight of a single satellite. A reduction in the weight of a separate satellite will be multiplied by a factor of four. Hence, reduction of satellite weight is likely to be more important than reduction of the incremental weight required to form the pallet/satellite combination.

In addition to possible reductions in satellite weight, alternative structural concepts could lead to a design wherein the satellite structure can be efficiently converted to the pallet/satellite combination structure without the dismounting and replacement of members.

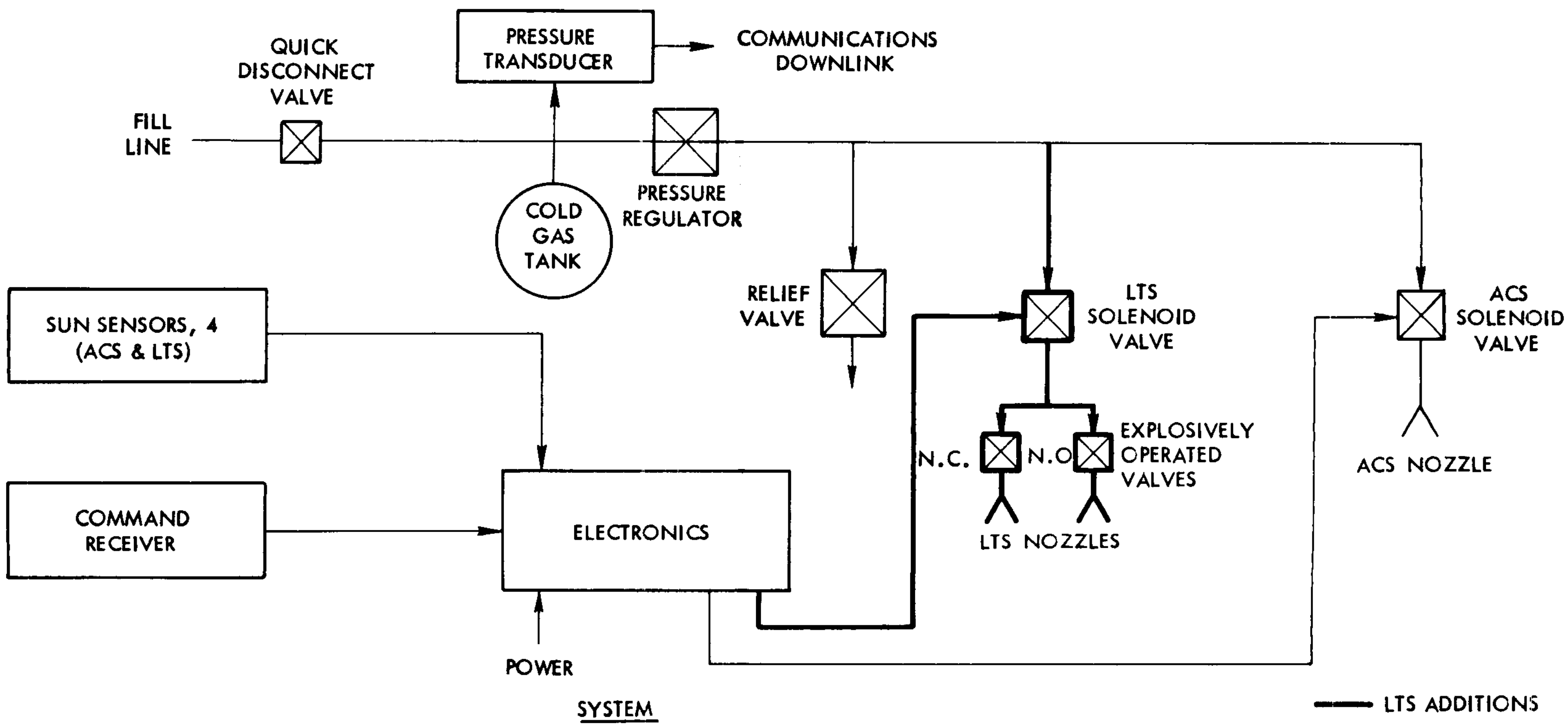
4.2.5 ATTITUDE CONTROL AND LATERAL THRUSTING SYSTEM

The integration of the attitude control system (ACS) and lateral thrusting system (LTS) for Deployment Scheme VI is shown diagrammatically in Figure 16. The systems share the cold-gas supply, the sun sensors and electronic circuitry for signal amplification and mode switching. The ACS part of the system is not significantly different from the Scheme I ACS, (except for a reduction in impulse requirements by a factor of about 2 due to reduced spin rate). Apart from the added functions to be accommodated by the command receiver and the subsystem electronics, the additional items required for the LTS implementation are those shown by the heavy lines in Figure 16; specifically, a solenoid valve, two explosively-actuated valves, two nozzles, and the associated electrical and pneumatic interconnections.

To avoid the generation of excessive disturbances torques the line of action of the lateral force must be shifted to match the shifts of the center of mass due to separation of the satellites from the assembly. This is accomplished as follows. When the LTS is first used only one nozzle will be active. This nozzle is appropriately located so that the line of action passes through the center of mass of the assembly at this time, i.e., after separation of the first satellite. After separation of the second satellite, the normally closed, explosively-actuated valve is opened. Both nozzles are then active and their thrust levels are balanced so that the line of action continues to pass through the assembly's center of mass. After separation of the third satellite, the normally open, explosively-actuated valve is closed, leaving only the second nozzle active. This nozzle is appropriately located so that the line of action continues to pass through the assembly's center of mass.

The directions in which lateral thrusts can be applied is limited by the directions of the view fields of the four solar sensors that are used to control the opening and closing of both the ACS and LTS solenoid valves. Investigation of the effects of this restriction on the efficiency of the execution of the orbital maneuvers indicates that the degradation of propellant utilization will be about two percent.

The preliminary estimates of the main engineering parameters of the system were based upon the achievement of a resolution in application of



SYSTEM

IMPULSE ~ 525 LB-SEC ACS
650 LB-SEC LTS
1175 LB-SEC TOTAL

PULSE DURATION ~ 0.050 SEC
ROTATION DURING PULSE ~ 18 DEG

ACS	
THRUST LEVEL	2.6 LB
MOMENT ARM	2 FT
ANGULAR PRECESSION PER PULSE	.05 DEG
TIME TO ROTATE 90 DEG	~ 1/2 HOUR
TOTAL NUMBER OF PULSES	~ 4,000

LTS	
THRUST LEVEL:	1 NOZZLE 2.6 LB 2 NOZZLES 5.2 LB
IMPULSE PER PULSE:	1 NOZZLE .13 LB-SEC 2 NOZZLES .26 LB-SEC
RESOLUTION OF ΔV	~ .01 METER/SEC
TIME FOR LARGEST ΔV	~ 60 MINUTES
TOTAL NUMBER OF PULSES	~ 5,000

Figure 16. ACS/LTS System

the velocity increment that is small enough to keep unplanned changes in tangential separation distance below about 300 km in 6 months. These estimates indicate that the design of the system will be compatible with the use of existing components.

4.2.6 ANTENNA DESIGN

Design considerations associated with Deployment Scheme VI indicate that the use of a collinear array antenna for the satellite's downlink transmission will be impracticable because of the excessive spacing that would be required between the satellites when they are mounted on the pallet. The cavity-backed slotted array antenna system depicted in Figure 17 provides a suitable substitute for the collinear array antenna and requires no protuberance beyond the height of the satellites. Thus, it permits minimum spacing between satellites.

This system consists of a set of antenna elements that are equally spaced about the periphery of the satellite and are energized through coaxial transmission lines in such manner that each element is equally illuminated and radiates in phase. Each element is a couplet comprised of a pair of stacked, boxed-in slot antennas. Each slot is half-wave resonant and is backed by a quarter-wave box to correct any shunt susceptance at the slot terminals, thereby making the slots nonreactive. The transmission is horizontally polarized (i.e., polarized perpendicular to the satellite's longitudinal axis).

The system is designed to radiate with reasonable uniformity in a plane perpendicular to the satellite's longitudinal axis (E-plane), and to be directional in the orthogonal plane (H-plane), providing a gain of 6.5 dBi at a frequency of about 2 GHz, i.e., equivalent to a fan-beam of 23 degrees. (For the Scheme VI design the satellite diameter of about 50 inches will be approximately 10 wavelengths.)

A high degree of uniformity (i.e., omnidirectionality of transmission in the E-plane power pattern) is necessary since the downlink power requirements will depend upon the lowest signal level received during a spin cycle. Hence, efficient power utilization requires the attainment of a near unity ratio between the minimum and the maximum of the E-plane power pattern. A high degree of E-plane uniformity can be obtained by careful control of the amplitude

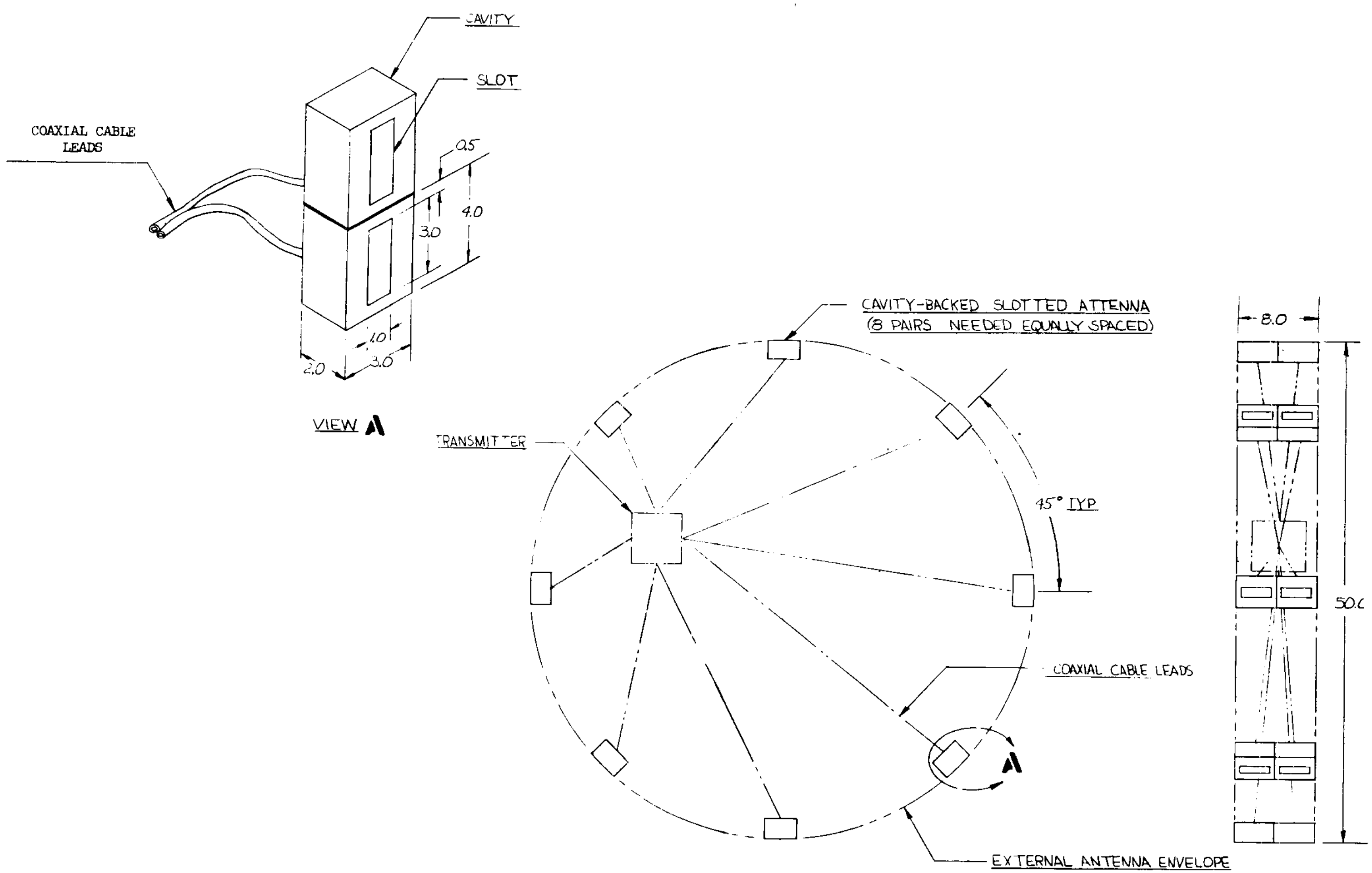


Figure 17. Cavity-Backed, Slotted Array Downlink Antenna System

and phase of each element, and by increasing the number of elements. However, increasing the number of elements ultimately results in a disproportionate increase in weight and cost. It is expected that six to eight elements will be near optimum and will provide satisfactory uniformity. For present purposes it is assumed that eight elements are used.

The H-plane power pattern, $P(\theta)$, can be expressed as the product of the pattern of two isotropic sources and the slot H-plane radiation pattern

$$P(\theta) = P_1(\theta)\, P_2(\theta)$$

where

$P_1(\theta)$ = pattern of two isotropic sources separated by d, the distance between slot centers

$P_2(\theta)$ = slot pattern

θ = elevation angle measured from satellite's lateral plane

If a half-wave spacing between slots is used, the H-plane power pattern function becomes approximately

$$P(\theta) = \frac{\cos^4\left(\frac{\pi}{2}\cos\theta\right)}{\sin^2\theta}$$

From this expression the 3 dB beamwidth is found to be equal to 24 degrees. A spacing slightly greater than a half-wavelength will yield a beamwidth of 23 degrees.

The gain of the antenna system, relative to isotropic transmission, is given by:

$$G = \frac{\eta}{\sin\left(\frac{1}{2}\theta_1\right)}$$

where η is the antenna efficiency and θ_1 is the H-plane beamwidth. With a 90 percent efficiency and a 23-degree beamwidth, a gain of 6.5 dBi will be achieved.

This downlink antenna system is to be shared by the telemetry and tracking transponder. With an appropriate frequency separation between the two transmissions and a suitable diplexing arrangement, it should be possible to keep the loss attributable to the sharing of the antenna below 0.1 dB.

Since the antenna elements are located at the periphery of the satellites, the spinning of the satellites will cause an oscillation of the translatory velocity of the elements. However, the doppler shift effects that are introduced by this velocity oscillation will not be more severe than the effects that were previously considered and found to cause no significant degradation of the ground reception.

Inasmuch as the satellite's uplink antenna, which is to be used for both command and transponder reception, requires a gain no greater than 2 dB, it can employ a simpler slotted array system, comprised of four, equally-spaced, single-slot elements. Though not omnidirectional this antenna provides broad angular coverage. Its approximately conical null region about the longitudinal axis can be tolerated since, except for a brief period shortly after injection into orbit when the spacecraft is still relatively close to the earth, the longitudinal axis will always be at a large angle to the satellite/earth line.

Only the pallet/satellite combination will be equipped with a tracking beacon. A single-slot element is adequate for the beacon antenna since a continuous beacon signal is not required.

One of the main problems involved in the implementation of the slotted array antenna is the design of a lightweight linkage system, which is complicated by the multiplicity of elements, the large distances between elements, and the need for precise control of relative phase and power level. It is felt that the weight of the antenna systems can be kept to about four or five pounds without departing from the use of currently available components. However, more detailed study of the linkage design will be required to firmly establish the system's weight.

4.2.7 POWER SYSTEM

The satellite power requirements and the system design are identical for the Configuration 1 and 2 designs. The subsystem requirement is for

17.5 watts and the solar cells provide 22 watts which allows for the various power losses. The solar cell area required is 2.6 ft^2 (projected) or 8.15 ft^2 (total) with a total weight, including support structure, of about 10.8 pounds.

The pallet power requirement has been discussed previously in the section on pallet/satellite integration. If there were no pallet/satellite integration, the requirement would be 475 watt-hours, which is considerably higher than the previous design due to the longer operating period of the satellite. As explained in the previous section, however, it is advantageous to combine the pallet/satellite power systems. The satellite can easily supply the pallet power, since the satellite instruments are not yet operating. The primary result of this integration will be the requirement for a modified power distribution system for the satellite which is integrated with the pallet.

4.2.8 THERMAL CONTROL CONSIDERATIONS

As for the previous multiple satellite design, Configuration 2 has been designed to utilize passive thermal control. Because of the different shape and size of Configuration 2, the thermal control finishes are different than Configuration 1. As in the previous design, solar cells are mounted around the periphery of the satellite, but in the previous design the top and bottom were painted white to achieve the desired temperature control. In the present configuration the surface area of the ends is substantially increased and the satellites would get too cold if painted white. It has been established through analysis that satellite temperature characteristics similar to the previous design can be achieved by painting 25 percent of the top and bottom surfaces with white paint and using polished aluminum surfaces for the remaining 75 percent. The white paint would be applied in one- to two-inch stripes to achieve a negligible temperature differential across the surface. Reference is made to Space-General Report 1089R-3, Figure 43, for the expected temperature profiles. In the calculation of the surface finishes required it was assumed that the inner annulus surface was insulated, which it must be to protect the satellite against the rocket exhaust.

Consideration has also been given to determination of the satellite temperature while still mounted on the pallet. The heat rejection areas are decreased for this case and the temperature of the satellites will increase. During this initial period of the mission, the average satellite temperature will increase by about 40°F to a maximum temperature of about 80°F. The inner satellites will have a somewhat higher temperature. As the satellites are deployed, the temperature will decrease to a maximum of about 40°F during the first part of their mission, i.e., when the sun is at apogee. When the sun is at perigee, the satellite will reach its maximum temperature, about 100°F.

4.2.9 WEIGHT STATEMENTS

Weight statements have been prepared for the Configuration 2 design and are shown in Tables 6 and 7. Table 6 shows a breakdown of the satellite weight with a total weight of 88.1 pounds. This is almost 5 pounds heavier than the previous satellite design. The total payload weight is shown in Table 7 and it is noted that the pallet weight, at 66.9 pounds, is substantially less than the previous pallet weight, almost compensating for the increased satellite weight. A summary of the total payload weight gives 435.3 pounds, which for an allowable weight of 446 pounds, allows a modest pad of 10.7 pounds.

4.2.10 DESIGN SUMMARY

The design of Configuration 2 has been guided by the basic study objective to maximize payload, volume and data rate allowable for scientific instruments, while providing adequate support subsystems and meeting all other constraints. These objectives have been accomplished to the depth of design possible within the scope of the study. Below is a summary of the system characteristics resulting from the design:

1. Instrument Allowances per Satellite

Weight	26 pounds
Volume	58000 in^3 (an order of magnitude greater than required)
Power	8 watts
Bit Rate	1050 bits/sec

2. Payload

Weight	435.3 pounds
Maximum Diameter	54.0 inches (shroud limitations)
Maximum Length	63.0 inches (shroud limitations)
Moment-of-Inertia Ratio	1.34 (booms stowed) 1.42 (booms deployed)

3. Satellite

Weight	88.1 pounds
Maximum Diameter	52.2 inches
Maximum Length	9 inches
Total Subsystem Power Available	18 watts
Boom Lengths	88 inches
Final Spin Rate	60 rpm
Component Temperature Range	20^{o} to $110^{o}F$
Moment-of-Inertia Ratio	> 1.4 (booms deployed)

Table 6

SATELLITE WEIGHT STATEMENT, SCHEME VI

Subsystem		Weight, Pounds
Science Instruments		26.0
IR Aspect Sensor		1.5
Power System		10.0
Solar Cells	4.0	
Battery	1.5	
Power Conditioner & Cabling	4.5	
Data Management System		27.0
Command Receiver/Decoder	3.0	
Data Processor	3.0	
Tape Recorder	8.0	
Transponder	7.0	
Transmitter	2.0	
Antenna	4.0	
Temperature Control		1.0
Structures and Mechanisms		22.6
Primary Structure	11.0	
Solar Array	6.8	
Magnetometer Booms (2)	2.0	
Mechanisms	2.8	
Total		88.1

Table 7

PAYLOAD WEIGHT STATEMENT, SCHEME VI

		Weight, Pounds
Pallet		
Data Management		2.5
Command Receiver/Decoder	.5	
Beacon & Antenna	2.0	
Power System		.5
Batteries	-	
Power Conditioner	.5	
Attitude Control and Lateral Thrusting Systems		38.9
Nitrogen Gas	16.8	
Pressure Reservoir	16.8	
Electronics	2.0	
Valves, Nozzles and Plumbing	2.5	
Solar Sensors (3)	0.8	
Thermal Control		-
Structure and Mechanisms		5.0
Solid Rocket(s)		19.6
Solar Aspect Sensor		.4
Total Pallet		66.9
4 Satellites @ 88.1 pounds each		352.4
Payload Adaptor		16.0
Total Payload Plus Adaptor		435.3
Payload Allowable		446.0
Pad		10.7

Section 5

DEPLOYMENT COMPARISON

5.1 ARRAY CHARACTERISTICS

Since the main function of multiple satellite deployment is to achieve a desirable array of intersatellite separation distances, the difference in the characteristics of the arrays that are obtained is an important factor in comparing Scheme I (Configuration 1) and Scheme VI (Configuration 2). Considering first, the three orthogonal components of the separation distance, the comparisons are as follows.

5.1.1 TANGENTIAL SEPARATION

For both schemes, the tangential separation distance arises from differences in lead time. Hence, the tangential separation distances will vary in proportion to orbital velocity and will be much smaller at large distances from earth than near perigee. In the case of Scheme I, an initial tangential separation distance of about 1,000 km will be obtained in the section of the orbit of most interest (near 12 R_e). This is a natural consequence of the spin-off separation. Subsequently, the tangential separation will grow to about 10,000 km in 6 months. This figure applies only to the growth due to the deliberate generation of orbital period differences which is required to obtain array non-coplanarity. Inadvertent differences in orbital period and other uncontrollable factors will introduce additional growth which could be as large as 15,000 km in 6 months. No capability for readjustment of the tangential separation distances is provided.

In the case of Scheme VI, the initial tangential separation distance is set by design and is selected to be about 1,500 km in the region of interest. Subsequent changes will be due mainly to differences in orbital decay rates. The growth in tangential separation due to this source will not exceed 1,500 km. The growth due to deployment errors is expected to be less than 300 km in 6 months. Since the pallet remains combined with the fourth satellite, its lateral thrusting system is available following deployment and can be used to readjust the tangential separation distance.

5.1.2 IN-PLANE NORMAL SEPARATION

The in-plane normal separation distance obtainable with Scheme I is severely restricted by the limitations of the spin-off separation. With a spin rate of 200 rpm, an initial in-plane normal separation distance of about 900 km can be obtained in the section of interest on the descending leg of the orbit; but on the ascending leg of the orbit, the initial in-plane normal separation distances will be negligibly small. As a result of orbital perturbation effects, the in-plane normal separation distance on the ascending leg will grow at a rate of about 200 km per month. On the descending leg the perturbations will have little effect on in-plane normal separation, since two of the satellites that are farthest apart in the in-plane normal direction remain relatively fixed while in-plane normal separation between the other two is reduced.

The initial in-plane normal separation distance obtained with Scheme VI depends only upon the propellant weight to be allowed for the in-plane normal separation maneuver. Since the initial spin rate does not affect the deployment it will be chosen at a value (about 100 rpm) appropriate for attaining the desired final rate after boom deployment. With the recommended maneuver, an initial 1,200 km in-plane normal separation distance will be obtained on the ascending leg, while a 1,800 km distance will be obtained on the descending leg. Subsequently, as a result of orbital perturbation effects, the in-plane normal separation distance on the ascending leg will increase at a rate of about 100 km per month, while the in-plane normal separation distance on the descending leg will decrease at about the same rate.

5.1.3 OUT-OF-PLANE SEPARATION

With respect to the out-of-plane separation distance there are no major differences between the two schemes. In both cases, the separation is obtained by firing axially-thrusting rockets so that the magnitude of the separation distance can be fixed by appropriate sizing of the rockets. In both cases, 1,500 km out-of-plane separation distances will be obtained in the section of interest on the ascending and the descending legs of the orbit. In neither case will there be much change in the out-of-plane separation distance subsequent to the deployment.

5.1.4 NON-COPLANARITY

While the attainment of each of the three orthogonal components of inter-satellite separation distance is necessary for non-coplanarity, it is not sufficient to assure non-coplanarity. In the case of Scheme I, at the outset, on the descending leg the non-coplanarity of the array will be poor because the separation vectors between satellite pairs will be nearly parallel; in addition to this factor, on the ascending leg non-coplanarity will be poor at the outset because of the absence of a significant in-plane normal separation distance. Non-coplanarity on the descending leg will first improve with the growth of the tangential separation, reaching a maximum at about two months after deployment; thereafter, continued growth of the tangential separation distance will tend to degrade the non-coplanarity of the array. On the ascending leg of the orbit, the improvement in non-coplanarity is slower because growth of the in-plane normal separation is required, in addition to the growth of the tangential separation distance. The long-term trend of the array non-coplanarity is speculative because of the uncertainties introduced by potential deployment errors and the absence of a capability for readjustment of any separation distance. It is most probable that the satellites will ultimately be strung out along the orbit in a very elongated array having poor non-coplanarity characteristics.

In the case of Scheme VI, strong non-coplanarity will be obtained at the outset on both legs and will be retained throughout the operational lifetime. The growth of tangential separation distance within a year will be small enough so as not to significantly degrade the non-coplanarity of the array and the capability to readjust the tangential position of the pallet/satellite combination can be used, if desired, to maintain near maximum non-coplanarity. The comparison of array characteristics is summarized in Table 8.

5.2 DESIGN IMPLEMENTATION

Configuration 1, shown in Figure 18, is applicable to Scheme I. In this design the satellites are of relatively small diameter and are joined in dumbbell-like pairs, with the dumbbell axes in line with the vehicle's longitudinal axis. The pairs are attached to the pallet spar in diametrically-opposed positions.

Table 8

COMPARISON OF ARRAY CHARACTERISTICS

	Scheme I	Scheme VI
Tangential Separation Distance	Approx. 1000 km initially, growing to at least 10, 000 km in 6 months; no readjustment capability.	1, 500 km initially; changes in the order of +1, 500 in 6 months; readjustment capability provided.
In-Plane Normal Separation		
Ascending Leg	Negligible initially; increases by about 200 km per month.	Initially 1, 200 km; increases by about 100 km per month.
Descending Leg	900 km initially (assuming spin rate of 200 rpm); little change due to perturbation.	Initially, 1, 800 km (independent of initial spin rate of about 100 rpm); decreases by about 100 km per month.
Out-of-Plane Separation	1, 500 km, initially, on both legs of orbit; little change with time.	1, 500 km initially, on both legs of orbit; little change with time.
Non-Coplanarity	Good non-coplanarity is not obtained initially. Non-coplanarity improves on descending leg to maximum at about 60 days and degrades thereafter; non-coplanarity on ascending leg is generally poor, improves late in operational life.	Good non-coplanarity is obtained on both legs of the orbit throughout operational life.

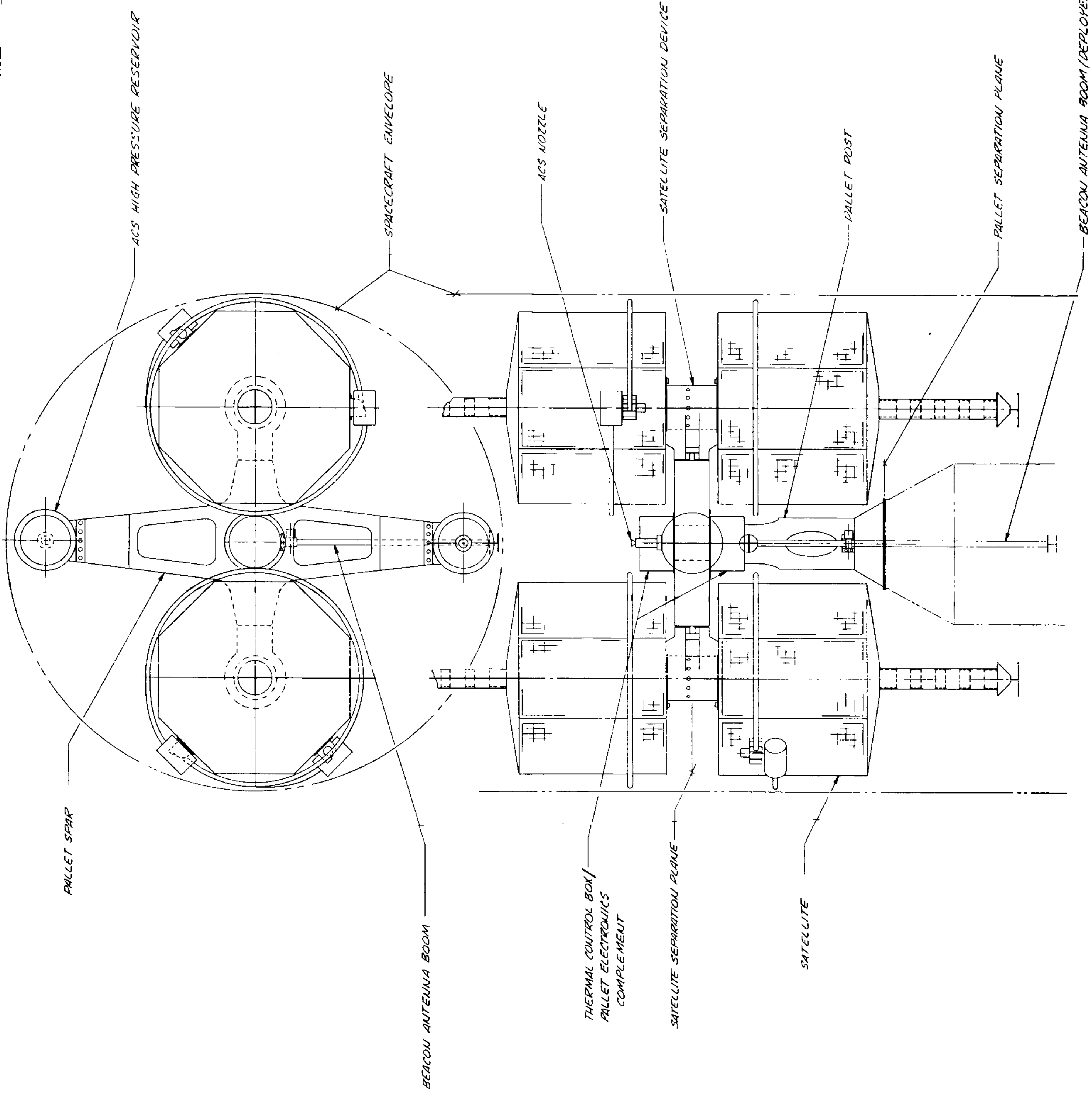


Figure 18. Pallet/Satellite Configuration 1

Configuration 2, which is applicable to Scheme VI, is shown in Figure 15. In this design the satellites are of relatively large diameter and are mounted in a stack extending along the longitudinal axis. The pallet is combined with one of the satellites to form a pallet/satellite combination. The attachment to the booster vehicle is made through this combination. One satellite is mounted below the combination while the other two are stacked above it.

In both cases, problems are encountered in attaining favorable moment-of-inertia ratios. For Configuration 1, a favorable moment-of-inertia ratio for the initial assembly is obtained by keeping the centers of mass of the satellite pairs as close as possible. Since the satellites are of relatively small diameter, the requirement for a low center of mass station will result in the need for tight packaging of the satellite subsystems. A favorable moment-of-inertia ratio could not feasibly be obtained for the satellite-pairs after separation from the pallet. To accept the unfavorable moment-of-inertia ratio it is necessary to spring the pairs apart very shortly after separation from the pallet. Also, the satellite precession dampers must be kept uncaged until after the pairs are sprung apart and a separate pallet precession damper is required.

For Configuration 2, a favorable moment-of-inertia ratio is obtained by minimizing the spacing between the satellites and by the use of the pallet/satellite combination. This precludes the use of a collinear array antenna for the downlink transmission which is replaced by a cavity-backed slotted array antenna at the cost of some additional weight for the more extensive RF linkage system that is required. However, with the attainment of a favorable moment-of-inertia ratio for the initial assembly, no further difficulty is encountered in obtaining favorable moment-of-inertia ratios for any of the assemblies that are involved in the deployment process. The large satellite diameter provides considerable shelf space for mounting subsystems, permitting inertial balance to be more readily obtained without the use of balance weights.

For both configurations, the allowable inertial tolerances are most stringent for the payload assembly mounted on the booster vehicle. These tolerances stem from the spin-stabilized booster vehicle's requirements and are the same for both configurations. However, for Configuration 1, relatively small tolerances must be held for all of the assemblies that are involved in the deployment process, while the tolerances for Configuration 2 are generally more liberal.

This is particularly true for the allowable tilt of the principal axes, which is about .1 degree for all Configuration 1 assemblies, while for Configuration 2 (except for the payload assembly) the tightest tolerance is about .35 degrees. Furthermore, for Configuration 1, the presence of inertial imbalances much greater than the allowable tolerances would result in serious degradation of the array; for Configuration 2 the adverse effects of inertial imbalances are mainly inconveniences that are correctable and do not endanger the mission.

The large diameter of the Configuration 2 satellites permits the stowage of telescopic booms, without the excessive number of segments that would be required if such booms were used with the smaller diameter satellites of Configuration 1. It is believed that telescopic booms will prove to be most desirable for deployment from the spinning satellites. For such booms, the possibility of hanging up or failure to lock is minimal since the centrifugal force acting to extend the booms persists until the boom is fully deployed even if its motion is strongly retarded by a viscous damper.

In the case of Configuration 1, thermal control of neither the separated satellites nor the initial assembly presents a very difficult problem. Although the heat rejection area will be reduced by a factor of 2 when the satellites are mounted on the pallet, the surface exposed to sunlight is also reduced by a considerable factor. Hence, the satellite temperatures on and off the pallet will be about the same.

In the case of Configuration 2, the satellite heat rejection area will also be reduced by a factor of about two when the satellites are still attached, but the surface exposed to sunlight will remain about the same. It is estimated that this will result in a bulk temperature of the initial assembly that is about $40^{\circ}F$ higher than the temperature of the separated satellites. More careful design will be required in this case to avoid large temperature gradients in the initial assembly.

The most difficult problem in implementing the Configuration 1 design is attainment of the required accuracy for the spin-off separation. A 1 millisecond error in the timing of the spin-off separation would result in an undesired growth in tangential separation distance at a rate of about 1,600 km per month. To avoid a gross distortion of the nominal array history, the maximum error should be kept below 1/2 millisecond. Standard separation mechanisms do not provide this order of timing accuracy. Experience with faster-acting separation mechanisms is limited; hence, their reliability is not established. Because of uncertainties regarding the separation mechanism and the complexities of the dynamic characteristics involved in the lateral separation of two elements from a spinning vehicle, main reliance must be placed upon extensive testing. These testing requirements would make the development of the spin-separation system a major item in the Multiple Satellite development program.

For Scheme VI there is no spin-off separation. All of the separations are axial and one-at-a-time and are, therefore, relatively simple and straightforward. The main drawback in this case is the need for the lateral thrusting system, which adds to the complication of the pallet equipment. However, the additional equipment required is minimized by integrating the LTS with the ACS in the manner illustrated in Figure 16. The additional equipment required for implementation of the LTS is actually smaller than the equipment that is required by Configuration 1 for implementation of the spin-off separation. The partial integration of pallet and satellite subsystems that is feasible with Configuration 2 and impracticable with Configuration 1 also results in a considerable reduction in the equipment required for implementation of the pallet's functions. This is demonstrated by the comparison of pallet equipment presented in Table 9.

The LTS operation is readily amenable to detailed analysis. Its components are generally on-the-shelf and their application is within the existing state-of-the-art. Hence, the test program associated with this equipment will be relatively routine. In contrast to the absence of a flight-proven precedent for the precision spin-off separation of Configuration 1, the use of a pulsed lateral thrusting system has been demonstrated by the Early Bird and Syncom satellites. The Multiple Satellite ACS/LTS system will be similar to the systems successfully used by these satellites.

While the ACS systems for the two configurations will be practically the same, the requirements on the Configuration 1 implementation will be more demanding because of the higher spin rate and because attainment of accuracy in the spin axis orientation is more critical. The comparison of design implementation is summarized in Table 10.

Table 9

COMPARISON OF PALLET EQUIPMENT

Scheme I Configuration 1	Scheme VI Configuration 2
ACS	ACS/LTS
Sun Sensors (4)	Sun Sensors (4)
Electronics	Electronics
Cold-Gas Supply: Tanks Fill Valve Relief Valve Pressure Regulator	Cold-Gas Supply: Tanks Fill Valve Relief Valve Pressure Regulator Pressure Transducer
Solenoid Valve (1)	Solenoid Valve (1)
Nozzle	Explosively-operated Valves (2)
Electrical and Pneumatic Connections	Nozzles (3)
	Electrical and Pneumatic Connections
Spin-Off System	
Sun Sensor	
Amplification and Firing Circuits	
Fast-Acting Separation Mechanism	
Beacon	Beacon
Command Receiver	Digital Solar Aspect Sensor
Deployable Antenna and Deployment Mechanism	Connectors to Satellite Power, Command and Telemetry Subsystems
Precession Damper and Uncaging Mechanism	Solid Rockets (2)
Solid Rocket	
Battery	

Table 10

COMPARISON OF DESIGN IMPLEMENTATION

Scheme I Configuration 1	Scheme VI Configuration 2
Small-diameter satellite mounted in diametrically opposed dumbbell assemblies	Large-diameter satellites stacked axially
Low satellite CM station required to obtain favorable moment-of-inertia ratio	Slotted array antenna required to obtain favorable moment-of-inertia ratio
Separated dumbbell assemblies do not have favorable moment-of-inertia ratio	All assemblies involved in deployment have favorable moment-of-inertia ratios
Limited envelope available for mounting satellite subsystems	Very large envelope available for mounting satellite subsystems
Tight inertial balance tolerances must be maintained, e.g., 0.1 deg on spin axis tilt	More liberal tolerance on inertial balance can be accepted, e.g., 0.35-degree spin axis tilt is allowable
Does not readily accommodate telescopic booms	Telescopic booms can be conveniently stowed
Control of temperature of all assemblies is relatively simple	Careful design to avoid large thermal gradients in initial assembly is required
Main Problem	
Development of Accurate Spin-Off Separation System	Development of LTS
Lacks state-of-the-art precedent.	Adds complication to the pallet system.
Experience with fast-acting separation mechanisms is limited.	Integration with ACS minimizes additional equipment required.
	Equipment is state-of-the-art.
Extensive test program required.	Test program is routine.
	Separations are simple.

5.3 WEIGHT STATEMENTS

Comparative weight statements for the satellites, pallet and total payload are presented in Tables 11 and 12. In view of the requirement for increased perigee altitude (lowest initial perigee altitude ≥ 830 km), an adequate payload capability cannot be obtained for either configuration without recourse to a velocity-kick at first apogee passage. Accordingly, the weight statements for both configurations have been based upon the assumption that such a kick will be applied and that injection into orbit will be at a perigee altitude of about 280 km.

Because of the additional cabling required (as a result of the greater dimensions) the satellite's power system will be slightly heavier for Configuration 2 (Scheme VI). The cavity-backed slotted array antennas and associated cabling used with Configuration 2 adds some weight to the data management system. Since the solid motors for obtaining the out-of-plane separation distance will be carried on the pallet, the solid motor weight is removed from the satellites; however, the larger satellite dimensions and the greater loads to be carried will increase the weight of the primary structure of the Configuration 2 satellites. Overall, an increased satellite weight of nearly five pounds is expected.

Configuration 2 saves some pallet weight by use of the pallet/satellite combination wherein command receiver and electrical power systems are integrated. Because of the higher spin rate, the Configuration 1 propellant requirements for the attitude reorientation maneuver are nearly as great as the combined ACS/LTS requirements of Configuration 2. The incremental weight of the structure of the pallet/satellite combination relative to the separate satellites is very small. This, along with the absence of the spin-off separation mechanism, separate precession damper, and deployable antenna results in a very substantial reduction of the weight assignable to the pallet structure and mechanisms.

In the Configuration 2 design the pallet must carry solid rockets: (1) for raising perigee altitude, and (2) for the out-of-plane separation maneuver, while in the Configuration 1 design the pallet requires only a rocket for raising perigee. However, since the Scheme VI maneuvers do not result in the reduction of perigee altitude for any of the satellites, while the Configuration I spin-off

Table 11

SATELLITE WEIGHT STATEMENT

Subsystem	Weight, Pounds: Scheme I (Configuration 1)		Scheme VI (Configuration 2)	
Science Instruments		26.0		26.0
IR Aspect Sensor		1.5		1.5
Power System		9.5		10.0
Solar Cells	4.0		4.0	
Battery	1.5		1.5	
Power Conditioner & Cabling	4.0		4.5	
Data Management		24.5		27.0
Command Receiver/Decoder	3.0		3.0	
Data Processor	3.0		3.0	
Tape Recorder	8.0		8.0	
Transponder	7.0		7.0	
Transmitter	2.0		2.0	
Antenna	1.5		4.0	
Temperature Control		1.0		1.0
Structures and Mechanisms		20.9		22.6
Primary Structure	6.8		11.0	
Superstructure				
Solar Array	6.8		6.8	
Magnetometer Booms (2)	2.0		2.0	
Mechanisms	2.8		2.8	
Solid Motor	2.5		-	
Total		83.4		88.1

Table 12

PAYLOAD WEIGHT STATEMENT

Subsystem	Weight, Pound Scheme I		Scheme VI	
Pallet				
Data Management		5.0		2.5
Command Receiver/Decoder	3.0			
Beacon & Antenna	2.0		2.0	
Power System		3.7		.5
Batteries	2.2		-	
Power Conditioner	1.5		.5	
Attitude Control and Lateral Thrusting Systems		35.2		38.9
Nitrogen Gas	15.7		16.8	
Pressure Reservoir	15.7		16.8	
Electronics	1.5		2.0	
Valves, Nozzles and Plumbing	1.5		2.5	
Solar Sensors (4)	0.8		0.8	
Thermal Control		1.0		-
Structure and Mechanisms		22.5		5.0
Solid Rocket(s)		14.5		19.6
Solar Aspect Sensor		-		.4
Total Pallet		81.9		66.9
4 Satellites		333.6		352.4
Payload Adaptor		16.0		16.0
Total Payload (Plus Adaptor)		431.5		435.3
Payload Allowable		446.0		446.0
Pad		14.5		10.7

separation results in a reduction of the perigee altitude for two of the satellites, a larger increase in perigee altitude is required for Configuration 1. Hence, for Configuration 1, a somewhat heavier rocket must be used to raise perigee altitude. This reduces the difference in solid rocket weights between the two configurations.

The reduced pallet weight of Configuration 2 nearly compensates for the increases in satellite weight, with the result that the total payload weight for Configuration 2 is about four pounds greater than the total payload weight for Configuration 1. In both cases, the total payload weight is within the capability of the booster vehicle.

5.4 OPERATIONAL REQUIREMENTS

The most obvious difference between the deployment operations for the two schemes is in the time required to complete the deployment. For Configuration 1, the deployment is completed in about 2-1/2 orbits, or approximately 5 days; the Configuration 2 deployment nominally requires 6-1/2 orbits, or about 13 days, and could take somewhat longer.

Because of the limitations on the pallet's battery, the completion of the Configuration 1 deployment within the specified maximum time is mandatory. Hence, although the deployment operations are fewer and shorter, they must be performed under the pressure of a hard deadline. On the other hand, the availability of solar power for the Configuration 2 pallet functions removes the pallet battery power constraint as a possible mission failure mode and permits the deployment operations to be conducted on a much less urgent basis.

In addition to the capability of postponing operations, if this becomes desirable, the Configuration 2 system provides data upon which intelligent operational decisions can be based. Access to the satellite's downlink permits the telemetering of ACS and LTS performance data. In particular, gas expenditure rates can be observed. If during ACS operation an expenditure rate greater than anticipated was noted this could be accommodated by reapportionment of the maneuver allocations, i.e., cutting down on the separation distances to be obtained. In the case of Configuration 1, a separate telemetry downlink from the pallet was not included. Even if the data were available, little could be done about an excessive ACS gas expenditure rate since there is no LTS supply from which gas could be reassigned.

The Configuration 2 attitude reorientation maneuver will not only be more secure with respect to gas supply but will also be aided by the digital solar aspect data. The attainment of a high accuracy in completion of the attitude reorientation maneuver is also much more critical to the success of the mission for Configuration 1 than for Configuration 2.

For Configuration 1, precise orbital determination is required at only two critical times in the deployment process: prior to the spin-off separation and prior to the firing of the solid rockets to obtain the out-of-plane velocity increments. In both cases, nearly a complete orbit is available from which to obtain tracking data for orbit determination. For Configuration 2, numerous orbital determinations will be required and in many cases only a comparatively small fraction of an orbit will be available. Frequently, the fraction of the orbit involved will be a section relatively close to earth where visibility is reduced and the orbit changes more rapidly. This poses a more difficult problem with respect to the collection and reduction of tracking data.

It is apparent that the Configuration 2 deployment operations will make greater demands on the ground stations. These demands will certainly extend over a longer period of time. The orbital maneuvers will entail several hours of ground command operations in comparison to the one-shot commands involved in the Configuration 1 deployment. More time will also be spent in collecting and processing tracking and aspect data. For the most part, the demands on ground station time for the deployment operations will not be much greater than for the subsequent routine collection of scientific data. However, to avoid the possibility of prolonging the Configuration 2 deployment operations, it may be necessary to obtain a priority with the ground stations for the Multiple Satellite mission during at least the early orbits. The comparison of deployment operations is summarized in Table 13.

Table 13

COMPARISON OF DEPLOYMENT OPERATIONAL REQUIREMENTS

Configuration 1 (Scheme I)	Configuration 2 (Scheme VI)
Deployment completed in 2 1/2 orbits; 5 days	Deployment completed in 6 1/2 orbits; 13 days
Deployment time limit imposed by pallet battery	No strict limit on deployment time
No downlink from pallet	Pallet downlink available; pallet operations monitored
No back-up for excessive ACS gas usage	LTS gas supply provides back-up for ACS
Accuracy of attitude reorientation is critical	Accuracy of attitude reorientation is not critical
Precise orbital determination required at only two critical times	Many orbital determinations are required
Nearly full orbit of tracking data available for orbital determinations	Orbit determinations must be made from data obtained in fraction of an orbit, sometimes from close-in data. More rapid tracking collection and reduction process required.
Ground stations tied up for shorter time. Relatively few ground commands required.	Several hours of ground commands required. Ground station priority may be required to avoid prolonging deployment

5.5 RECOMMENDATION

Based on studies conducted to date, it is concluded that the foregoing comparison indicates Configuration 2 to be preferable for the Multiple Satellite mission. The reasons for recommending Configuration 2 are, in summary:

- Configuration 2 will permit the attainment of an array of intersatellite separation that is far superior to the best array that could possibly be attained by Configuration 1.

- The implementation of Configuration 2 is state-of-the-art and is supported by flight-proven precedents. Although the components are state-of-the-art, and development is entirely feasible there is no known precedent for the development of the precision spin-off separation system required by Configuration 1.

- The payload weights for the two configurations are comparable.

- While the deployment operations for Configuration 2 are more extensive, the absence of a hard deadline and the inherent back-up and corrective features that are provided suggest greater assurance of their successful completion.

Section 6

AVAILABLE SATELLITE SURVEY

A brief survey of existing satellite structures has been initiated to determine the possibility of using an existing structure for the multiple satellite. Based upon this survey, none of the existing satellite structures is applicable to the current multiple satellite design alternatives. This initial investigation has been based only upon the compatibility of dimensional properties, moment-of-inertia characteristics, and center of gravity location of the existing spacecraft relative to the multiple satellite design alternatives. There are two multiple satellite designs currently under consideration and the dimensions of these are shown as the first two satellites of Table 14. The Reference 1 design corresponds to the Deployment Scheme I configuration which is deployed by spin-off. The Reference 2 design corresponds to the Deployment Scheme VI configuration which is deployed by a lateral thrusting system.

Most of the recently orbited satellites were considered in this investigation, including those mentioned in the Statement of Work, i.e., S^3, Pioneer, OGO, IMP, and ATS. These satellites, plus others which approach the dimensional requirements of one of the reference designs, are illustrated in Table 14.

The Reference 2 design has many system advantages for the Multiple Satellite program and may eventually be the selected design, as explained in Section 5.5. There is no other existing satellite, however, which even approaches this configuration and which could be effectively used for this design.

For the Reference 1 design, several existing structures must be considered more closely. In particular, these are the S^3, IMP, OV3, SYNCOM, BEACON, SOLRAD, LES 1 and INJUN. These are all shown in Table 14. The other satellites shown in Table 14 are obviously too large. The IMP,

Table 14

MULTIPLE SATELLITE SPACECRAFT SURVEY - EXTERNAL DIMENSIONAL CONSIDERATIONS

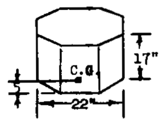
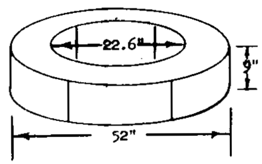
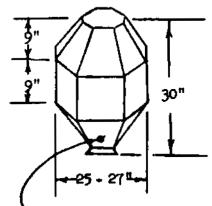
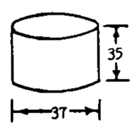
SATELLITE	DESCRIPTION	DIMENSIONS	REMARKS
<u>Ref. 1</u> Multiple Satellite Deployment Scheme I Configuration	Axially symmetric octagon with flat forward and aft closures. Sides except forward and aft closures are covered with solar cells.		1. To maintain required power the projected area of the sides must not be decreased. 2. The diameter may not be greater than shown in order to attach to the pallet and fit within the shroud; it may be smaller providing there is room to mount the equipment and the moment-of-inertia ratio is maintained favorable i.e. the C.G. remains as shown. 3. The height may be increased or decreased providing the pallet and satellites can fit within the shroud and the moment-of-inertia ratio remains favorable.
<u>Ref. 2</u> Multiple Satellite Deployment Scheme II Configuration	Axially symmetric octagon (may eventually have more sides). Doughnut shaped with flat forward and aft closures. Sides except forward and aft closure are covered with solar cells.		Little dimensional change is allowed to meet the moment-of-inertia, shroud and solar cell power limitations.
1. SMALL STANDARD SATELLITE (S^3)	Axially symmetric satellite having octagonal center section and truncated ends.		1. Obviously unacceptable for Ref. 2 design. 2. Unacceptable for Ref. 1 design because: a. Diameter too large to mount on pallet and fit within shroud. 52" is maximum diameter allowed - two satellites could possibly fit side by side but there would be no room for mounting on the pallet. b. The height is much too large to maintain favorable moment-of-inertia ratio on pallet; the conical configuration prevents weight from being located near lower portion of satellite (see shadowed section of sketch) which is required to achieve favorable moment-of-inertia on pallet. Note location of required C.G. 
2. PIONEER	Axially symmetric cylindrical satellite with flat forward and aft closures. Solar cells mounted around cylindrical surface except for bellyband.		1. Obviously too large for Ref. 1 design. 2. Unsuitable shape for Ref. 2 design.

Table 14 (Continued)

MULTIPLE SATELLITE SPACECRAFT SURVEY - EXTERNAL DIMENSIONAL CONSIDERATIONS

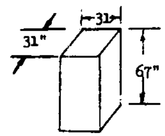
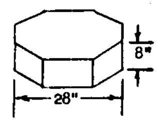
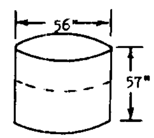
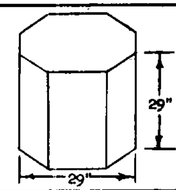
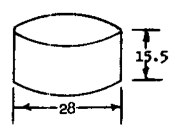
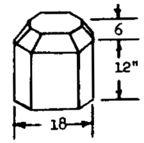
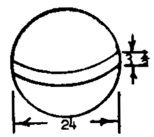
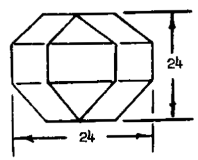
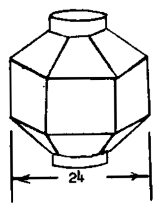
SATELLITE	DESCRIPTION	DIMENSIONS	REMARKS
3. ORBITING GEOPHYSICAL OBSERVATORY	Aluminum box-like body. Solar panels provide power.		Completely unsuitable for either design.
4. IMP	Main structure is an octagon with forward and aft closures. Solar paddles provide power.		1. Unacceptable for Ref. 1 design because: a. The 28" diameter is too large; it would not be possible to mount the four satellites within the shroud. b. The projected area on which solar cells could be mounted is too small. 2. Obviously unacceptable for Ref. 2 design.
5. APPLICATIONS TECHNOLOGY SATELLITE (ATS)	Axially symmetric right cylinder. Cylinder covered with solar cells.		Much too large.
6. OV3	Right octagonal cylinder. Solar cells mounted on all sides.		1. Both diameter and length too large for Ref. 1 design. 2. Unsuitable shape for Ref. 2 design.
7. SYNCOM	Right cylinder faced with solar cells.		1. Comes close to being acceptable for Ref. 1 design but diameter is too large for mounting four satellites within the shroud. 2. Unacceptable for Ref. 2 design.

Table 14 (Continued)

MULTIPLE SATELLITE SPACECRAFT SURVEY - EXTERNAL DIMENSIONAL CONSIDERATIONS

SATELLITE	DESCRIPTION	DIMENSIONS	REMARKS
8. BEACON EXPLORER	Octagonal main body with truncated cone. Solar panels and solar cells mounted on the truncated cone provide power.		1. This satellite structure could become acceptable for the Ref. 1 design if the power requirements were reduced and some of the equipment made smaller. But based on current requirements it is too small. 2. Unsuitable for Ref. 2 design.
9. EXPLORER 30 (SOLRAD 8)	Two hemispheres separated by an equatorial band. Six 11" circular solar panels mounted symmetrically on hemispheres.		1. Unsuitable for Ref. 2 design. 2. Diameter possibly too large for Ref. 1 design (requires further design investigation). The shape, narrowing toward bottom may preclude location of the satellite C.G. low enough for the pallet assembly to be stable.
10. LINCOLN EXPERIMENTAL SATELLITE (LES 1)	Polyhedron with 18 square faces and 8 triangular faces. Square panels covered with solar cells. Sensors on triangular panels.		Comments same as Satellite 9.
11. EXPLORER 25 (INJUN 4)	Aluminum shell with 40 sides, 30 sides covered with solar cells. Cylindrical tube on contained Explorer 24 ballon.		Comments same as Satellite 9. Elimination of tube on top will require modification.

OV3, and SYNCOM all have diameters which are larger by six inches or more than the Reference 1 design. Because of the pallet mounting arrangement and the shroud design limitations, these diameters are unacceptable and the satellites must be rejected. In addition, only the SYNCOM satellite has an acceptable height. The Small Standard Satellite (S^3) has a diameter approximately one-inch smaller than the three satellites mentioned previously; this is also too large to be acceptable. In addition, it has excessive height to maintain a favorable moment of inertia ratio, especially while the four satellites are mounted on the pallet. That is, the major part of the satellite weight must be located near the bottom of the satellite which has a reduced cross-sectional area making it impossible to meet the C.G. location requirement.

The Beacon Explorer satellite is too small. It would only be acceptable if the power requirements and payload were decreased.

The SOLRAD, LES I and INJUN satellites are all somewhat similar. They are all 24 inches in diameter and "rounded" in shape. There is more similarity between these satellites than between their designs and the Reference 1 design, yet separate satellite structures were constructed for these three satellite functions, implying that it would be best to develop a separate multiple satellite structure. Even though these satellites have dimensional similarities to the Reference 1 design and could provide the required power with body-mounted solar cells, they have serious disadvantages. First, the diameter is somewhat larger which may incur a difficult, if not impossible, pallet mounting design problem. Secondly, the height is too great which could, depending upon internal payload arrangement, result in satellite moment-of-inertia ratio problems. Whereas the single-satellite moment-of-inertia ratio could probably be developed favorably, the design of a favorable moment-of-inertia ratio for the pallet/satellite combination is questionable. This requires that the payload weight be concentrated near the bottom of the satellite. But, for these satellites the volume available decreases toward the bottom. The center-of-gravity of each satellite must be within five inches of the bottom of the satellite to maintain acceptable pallet/satellite moment-of-inertia ratios. This is virtually impossible for any of the three satellite designs. Therefore, these satellites were also rejected.

Additional satellites which have been considered and the reasons for their rejection are listed in Table 15.

In summary, unless the multiple satellite design requirements change substantially from either of the two reference designs given in Table 14. it can be stated that none of the existing satellite structural configurations considered is acceptable based upon dimensional and moment-of-inertia constraints. If the multiple satellite designs do change substantially from the reference designs (which seems unlikely), further survey effort may be required to determine the applicability of existing satellites to the multiple satellite program. In this case, it may be necessary to go beyond dimensional, moment-of-inertia and center-of-gravity location considerations and to evaluate other pertinent factors, such as structural design, loading, mounting, etc.

Table 15

ADDITIONAL SATELLITE STRUCTURES

Satellite	Description	Remarks
1. Biosatellite	Re-entry structure maximum diameter - 40 inches	Much too large and of unsuitable design
2. Early Bird	Cylinder 28 inches in diameter, 20" high	Too large for Ref. 1 design; shape and size unsuited for Ref. 2 design
3. Environmental Research Satellite		
(ERS 16)	Octahedron 9" on a side	Too small
(ERS 17)	Octahedron 11" on a side	Too small
4. Environmental Survey Satellite (ESSA)	18 sided cylinder-like polygon, 42" in diameter and 22" high	Too large
5. Explorer 20	Cylinder with truncated cone on top and bottom 26" in diameter and 46" high	Too large, especially in height for Ref. 1 design; unsuitable size and shape for Ref. 2 design
6. Explorer 26	Octagonal planform atop a truncated cone 28" in diameter and 17" high	Diameter too large for Ref. 1 design; shape and size unsuitable for Ref. 2 design
7. GEOS	Octagonal aluminum shell 48" across flats, 32" high	Too large
8. Initial Defense Communications Satellite	Symmetrical polyhedron with 24 faces, 32" high and 36" in diameter	Too large for Ref. 1 design; shape and size unsuited for Ref. 2 design
9. INTELESAT	56" in diameter, 26" high	Too large
10. Lincoln Experimental Satellite (LES 4)	Ten-sided polyhedron 33.5" in diameter, 36" high	Too large for Ref. 1 design; shape and size unsuited for Ref. 2 design

Table 15 (Continued)

ADDITIONAL SATELLITE STRUCTURES

Satellite	Description	Remarks
11. NIMBUS	--	Much too large
12. Nuclear Detection Satellite (VELA)	Polyhedron 54" in diameter	Too large
13. Orbital Astronomical Observatory (OAO)	--	Much too large
14. Orbiting Solar Observatory (OSO)	Wheel section 44" in diameter, 9" high	Diameter too small for Ref. 2 design
15. OSCAR	7 x 12 x 17 box	Too small and unsuitable for spinning spacecraft
16. OV1	27" diameter cylinder, 55" long with hemispherical forward end	Too large and the shape is unsuitable
17. OV2	Main body 23" square and 24" long	Not well suited for spin stabilized vehicle
18. Pegasus	Open truss supporting 2 large "wings"	Completely unsuited
19. Relay	Octagonal prism 29" diameter at broad end, 33" high	Too large for Ref. 1 design and shape and size unsuited for Ref. 2 design
20. SECOR	9 x 11 x 14 box	Too small and unsuitable for spinning spacecraft
21. Tetrahedron Research Satellite (TRS)	Tetrahedron $6\frac{1}{2}$" on a side	Too small
22. TIROS	Cylindrical 18 sided polygon 42" in diameter 22" in height	Too large for Ref. 1 design and shape and size for Reference 2 design
23. Miscellaneous Explorer, Echo, Pageos, etc.	--	Unsuitable in design